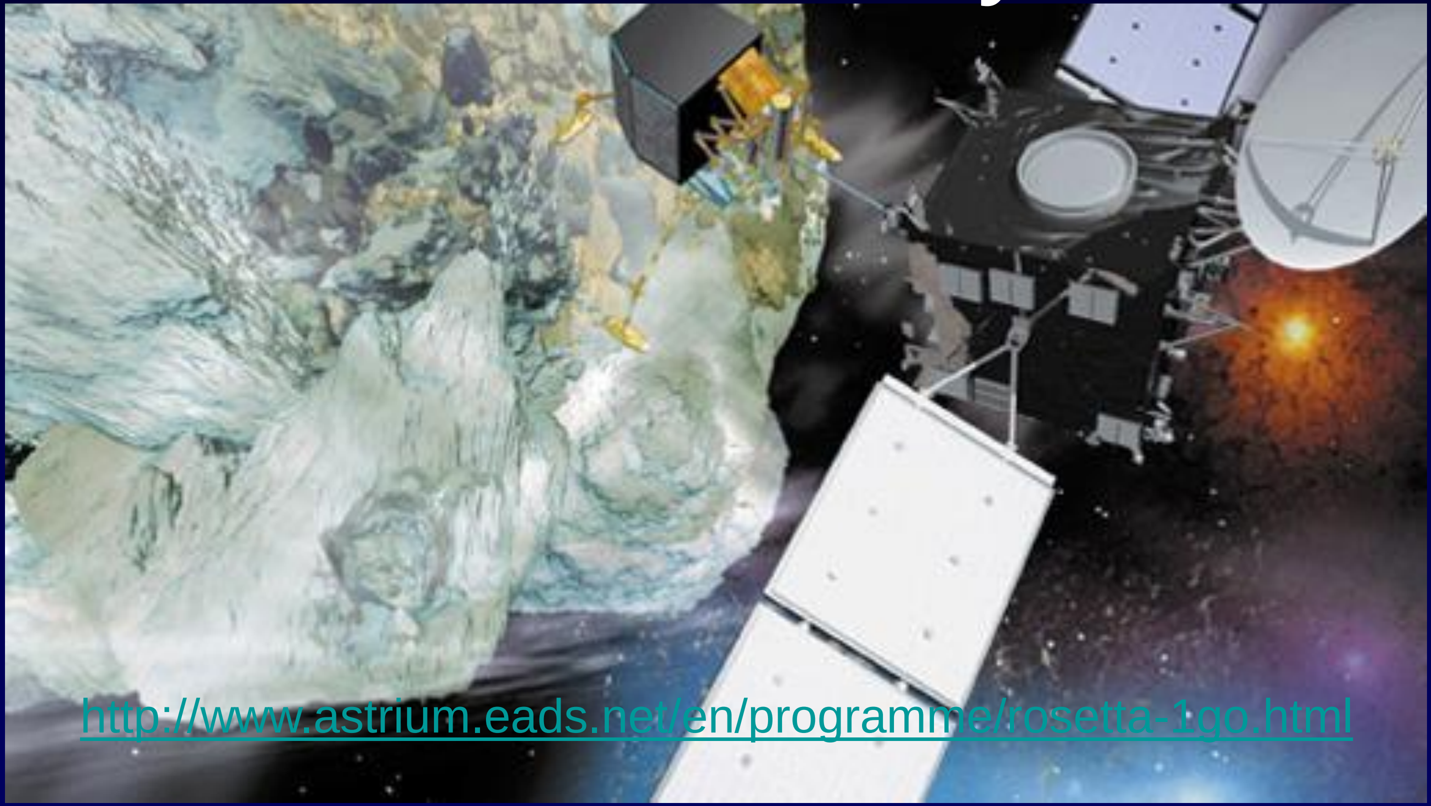
Satellite & Space Systems – Electrical Power Systems



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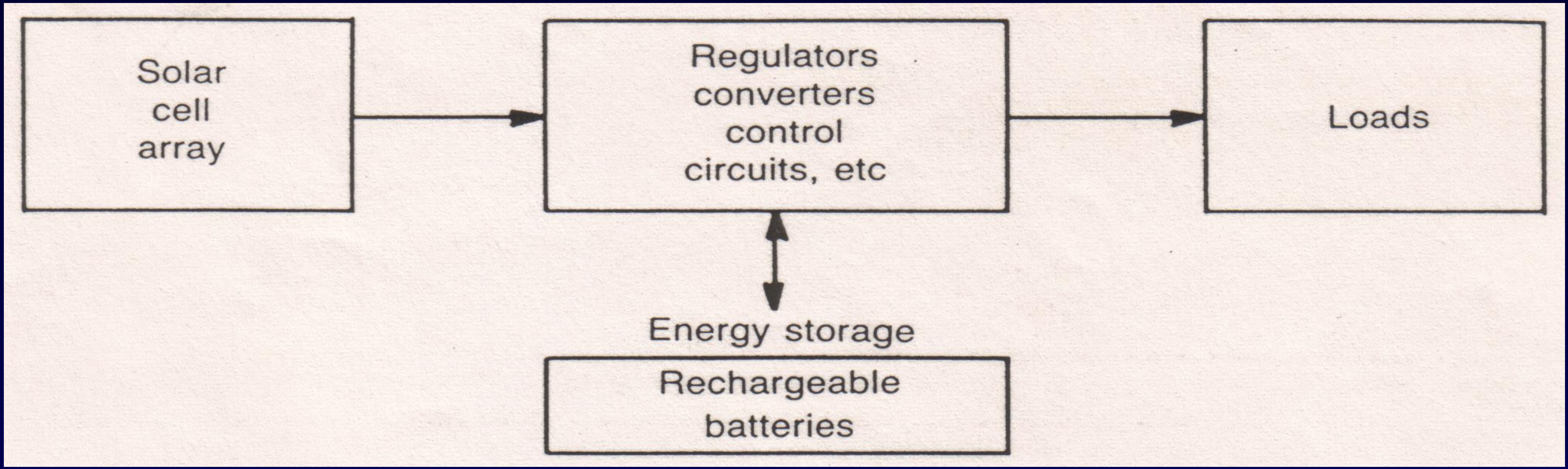
MSc/MEng – Satellite and Space Systems

UNIVERSITY OF SUSSEX
SCHOOL OF ENGINEERING & INFORMATICS

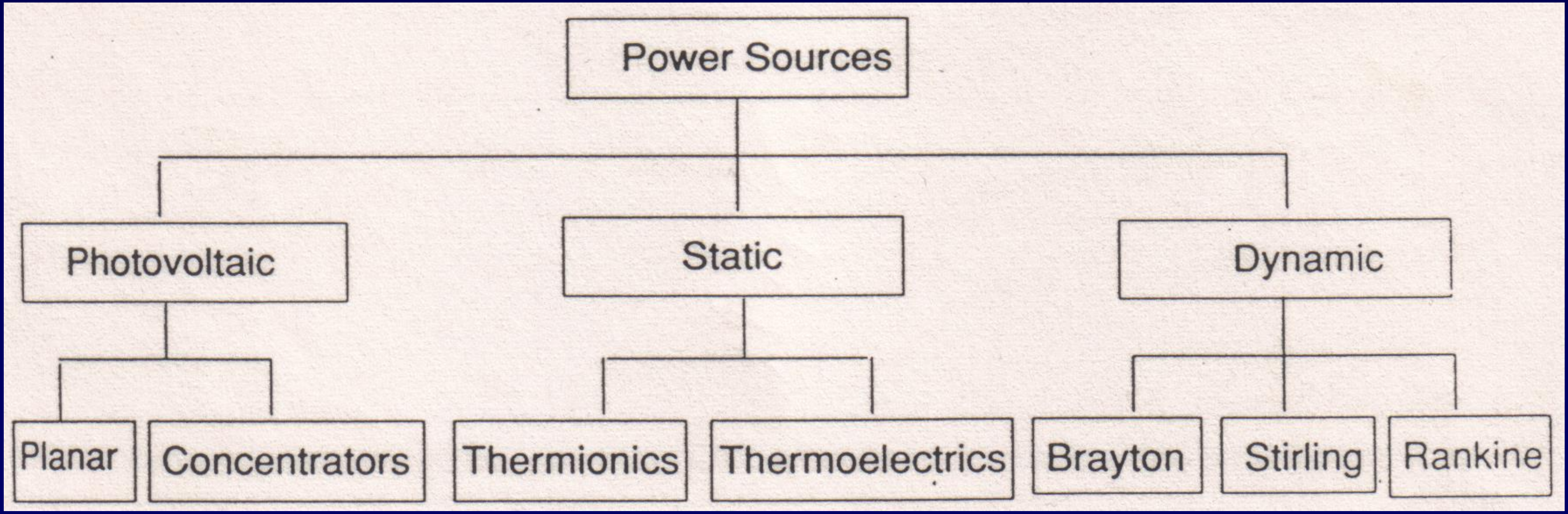
25TH APRIL 2017

Power2 Systems

Power System = Power Generation + Storage + Conditioning + Utilization



Types of Power Source:



Energy Sources and Operational Scenario

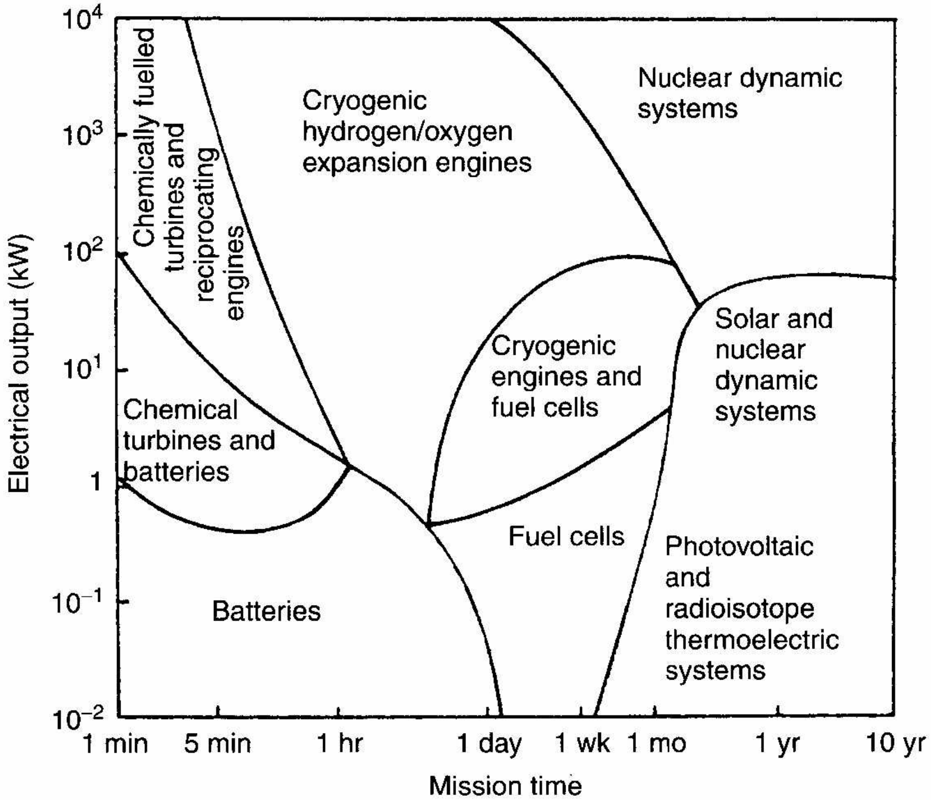


Figure 10.1 Power outputs: mission duration relationship between energy source and appropriate operational scenario [2] (From Angrist, S. W. (1982) *Direct Energy Conversion*, 4th edn, Copyright Allyn and Bacon, New York)

Typical Power System Block

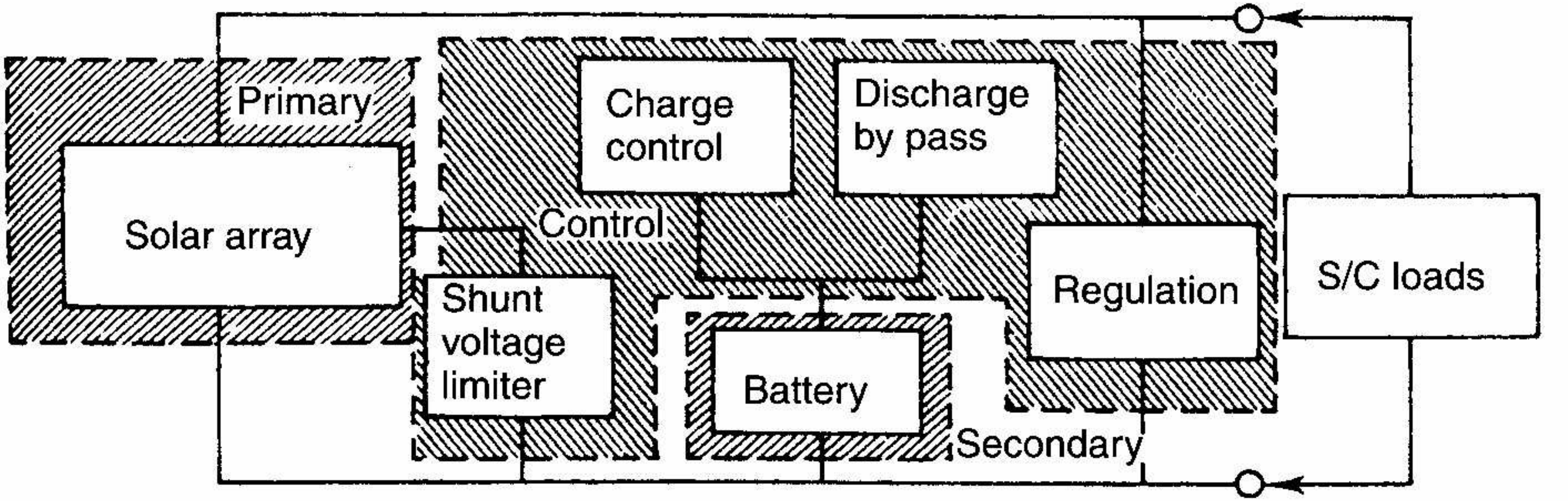
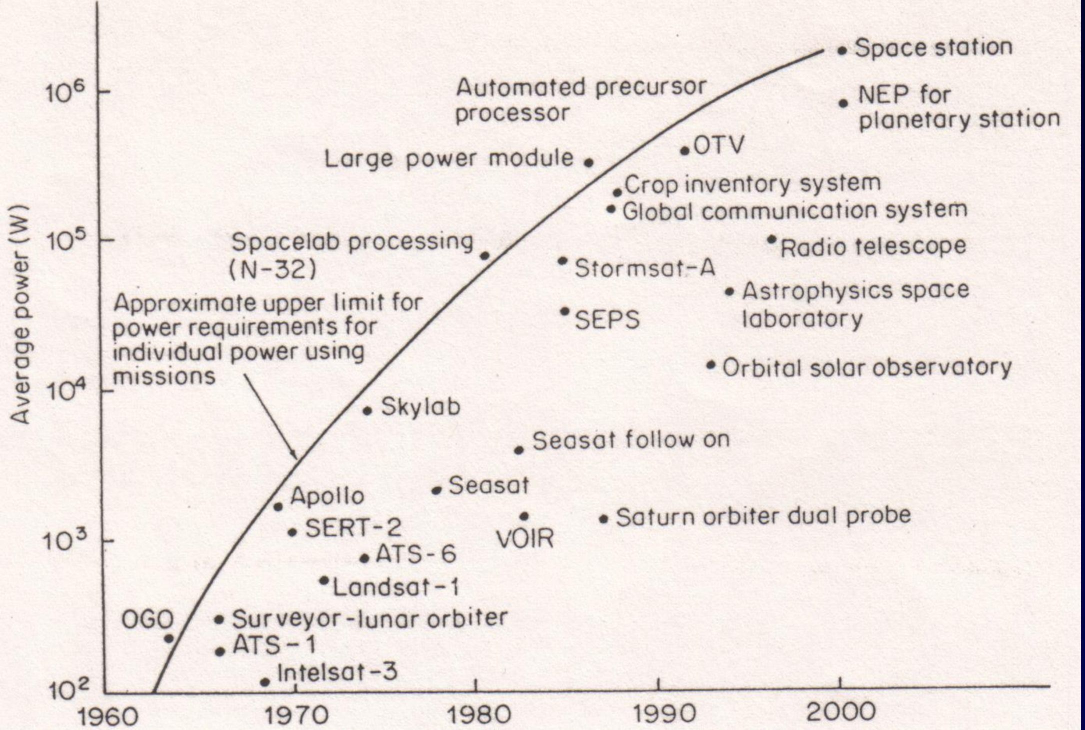
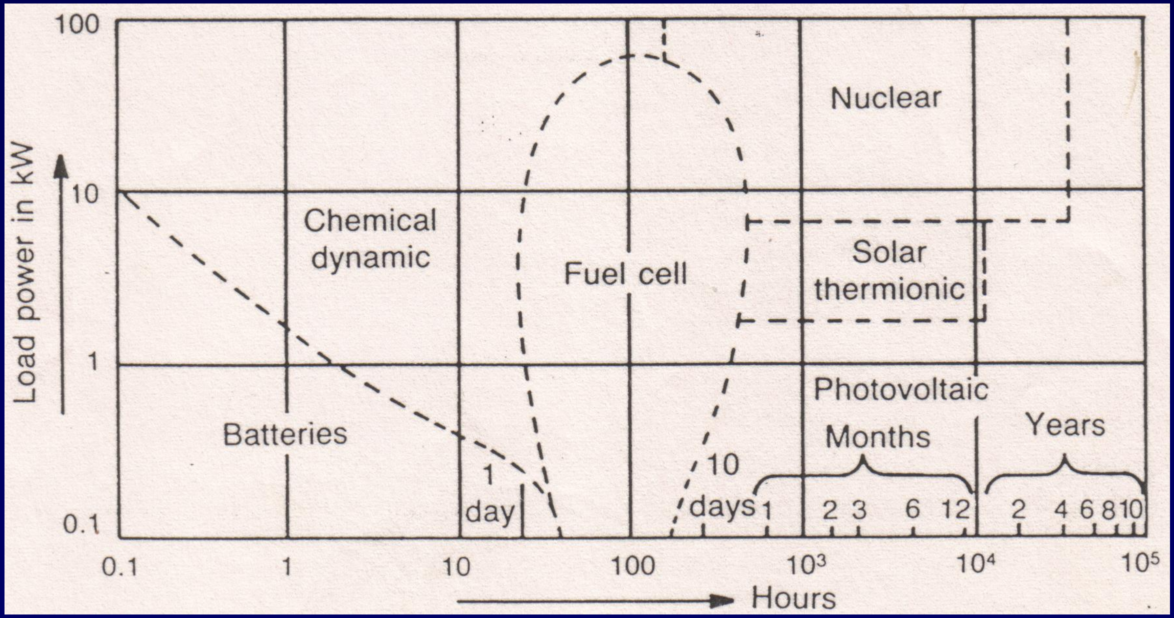


Figure 10.2 Schematic of typical spacecraft power system block elements

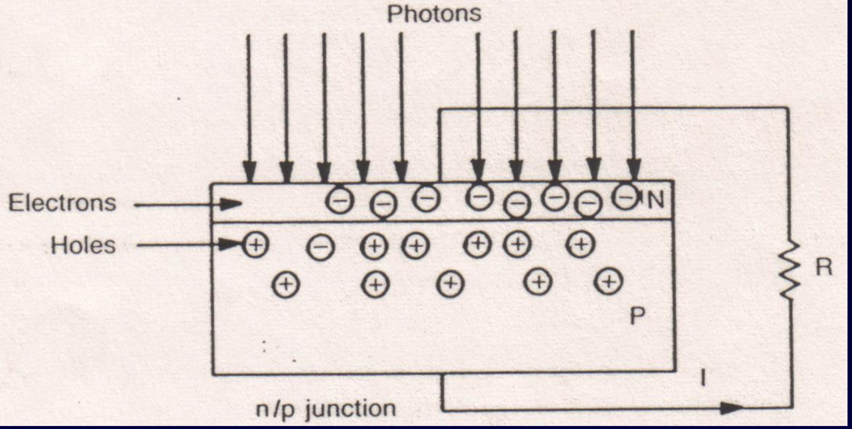
Power Demand Evolution

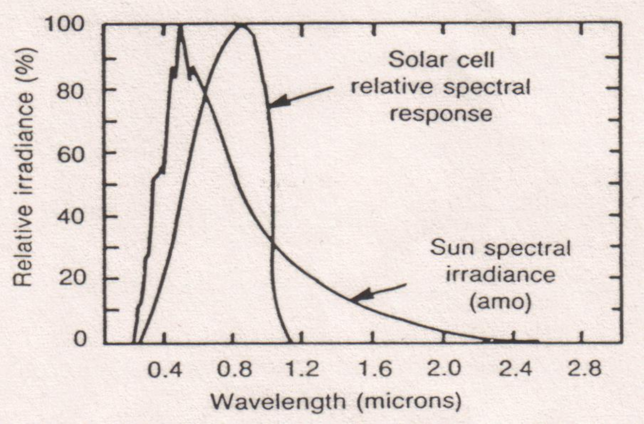




Power delivery - duration as a function of type

Solar Cells – Most Common Source

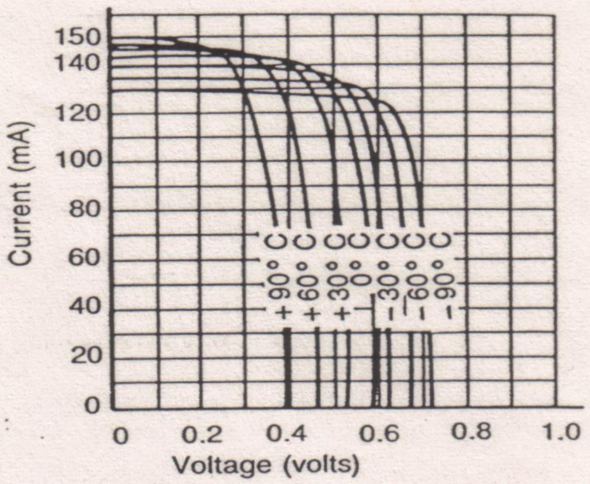




Photons create electron-hole pairs near junction of N & P types of semiconductor electron-hole pairs diffuse with
e → N region   and hole → P region
→ resistance decreases,
current flows in external load R.

Spectral response is biased towards the red side of the peak in solar radiation.

Current-Voltage curves →
Ideal, maximum power, bias for max   I x V,
→bias at knee of curve,
but temperature dependant
→ needs tracking to keep on knee



Parameter	Commercial Cell	COMSAT Cell
Collection efficiency	71%	79%
Reflection losses	9.5%	4.9%
Open circuit voltage	0.580 V	0.595 V
Series resistance	0.17 ohm	0.05 ohm
Conversion efficiency	10.4%	14%

← Efficiencies low typically 10-15%.

Some New Triple junction cells >40%

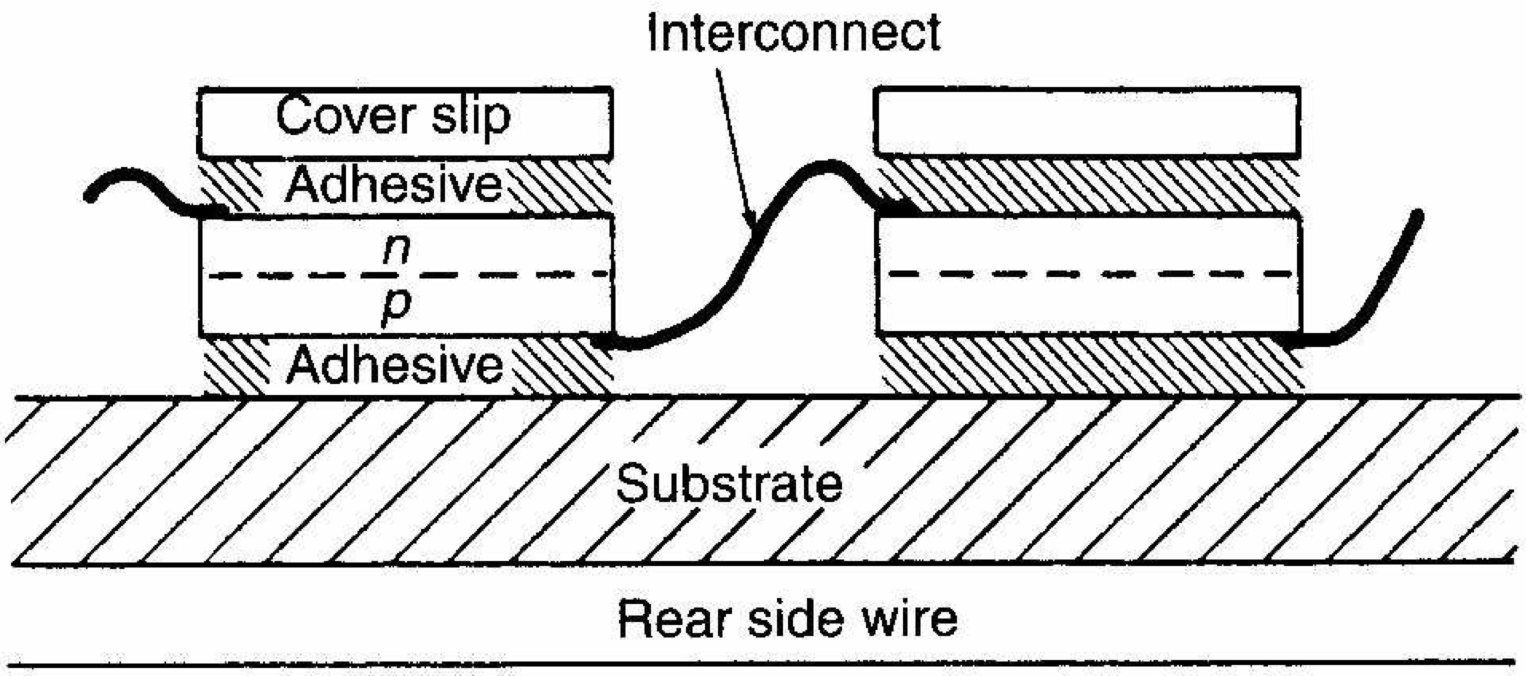


Figure 10.3 Schematic of a typical solar cell assembly

A diagram of a solar cell

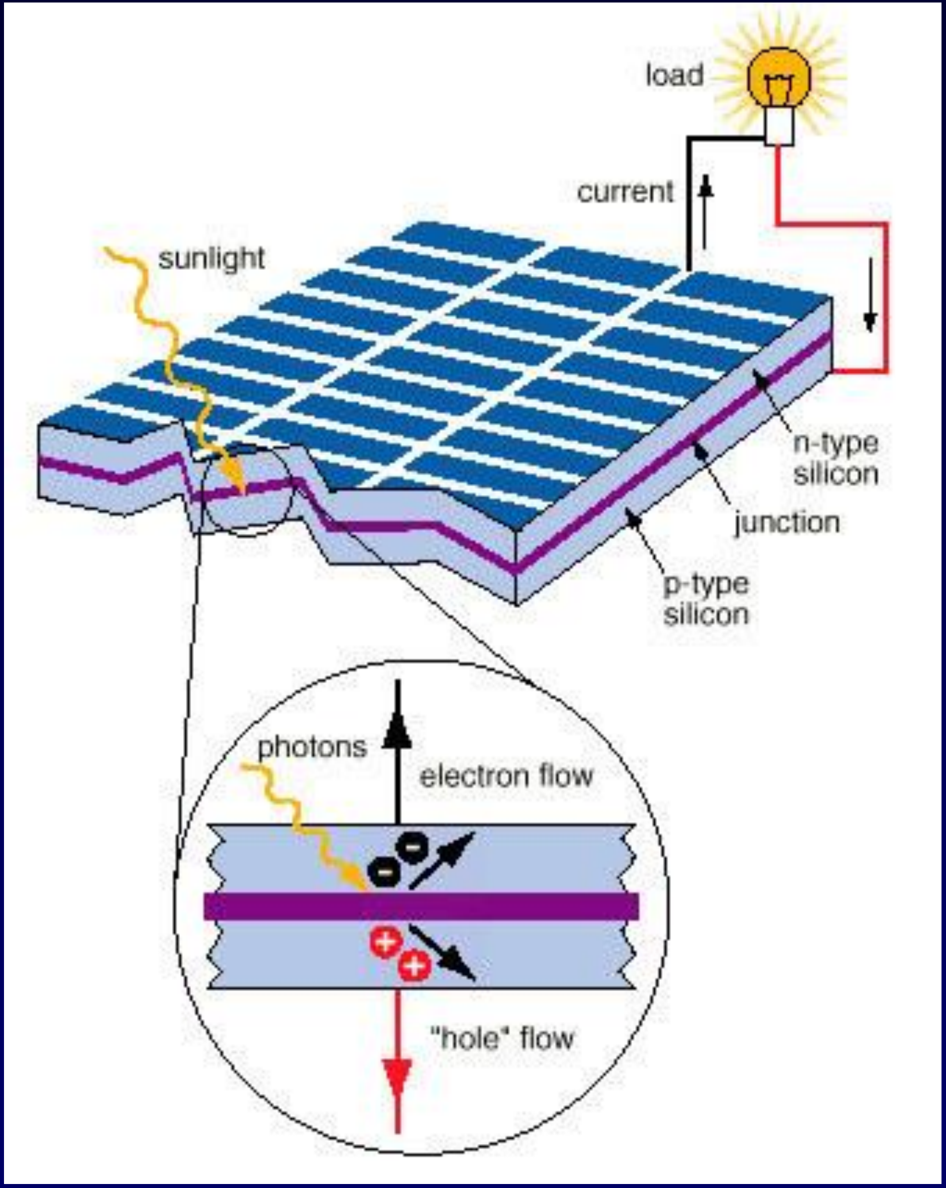


Table 10.1 Properties of semiconductor materials

Material	Band gap (eV)	Maximum wavelength (μm)
Si	1.12	1.12
CdS	1.2	1.03
GaAs	1.35	0.92
GaP	2.24	0.554
CdTe	2.1	0.59

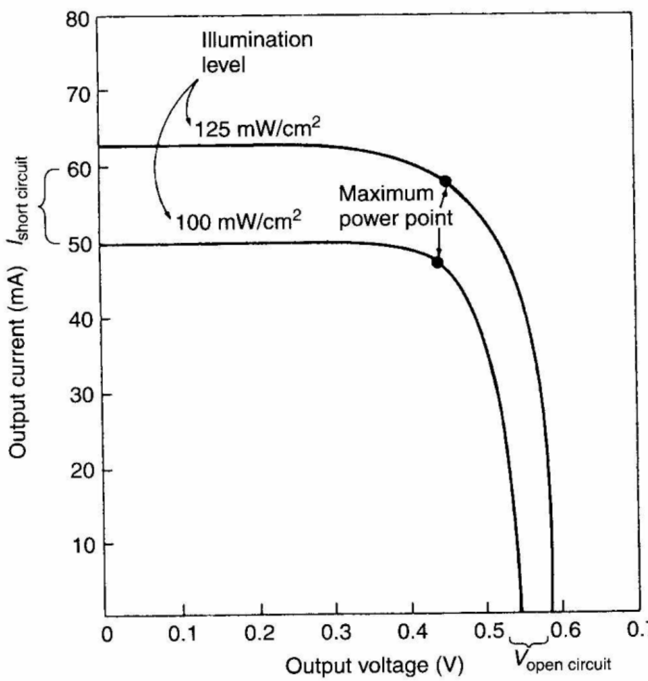


Figure 10.4 Current–voltage characteristic for a typical solar cell. The short-circuit current is dependent upon both the illumination level and the size (area) of the cell (From Angrist, S. W. (1982) *Direct Energy Conversion*, 4th edn, Copyright Allyn and Bacon, New York)

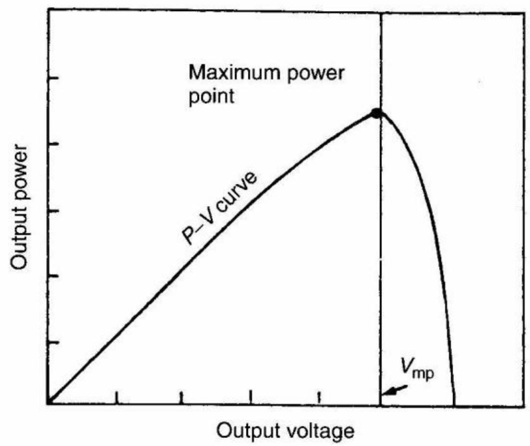


Figure 10.5 Power–voltage characteristic for a typical solar cell (From *Solar cell array design handbook* by Rauschenbach, H. S. Copyright © 1980 by Van Nostrand Reinhold. All rights reserved)

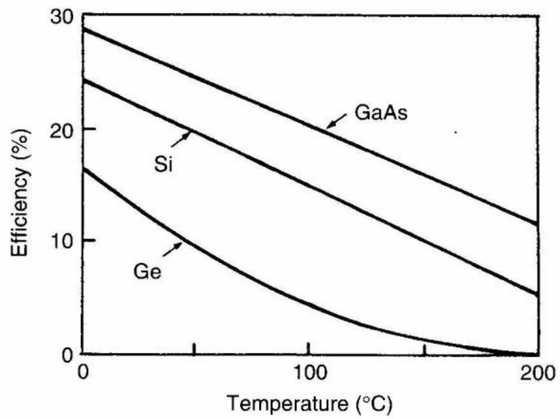


Figure 10.6 Theoretical cell efficiency as a function of temperature for three semiconductor materials (From *Solar Cell Array Design Handbook* by Rauschenbach, H. S. Copyright © 1980 by Van Nostrand Reinhold. All rights reserved)

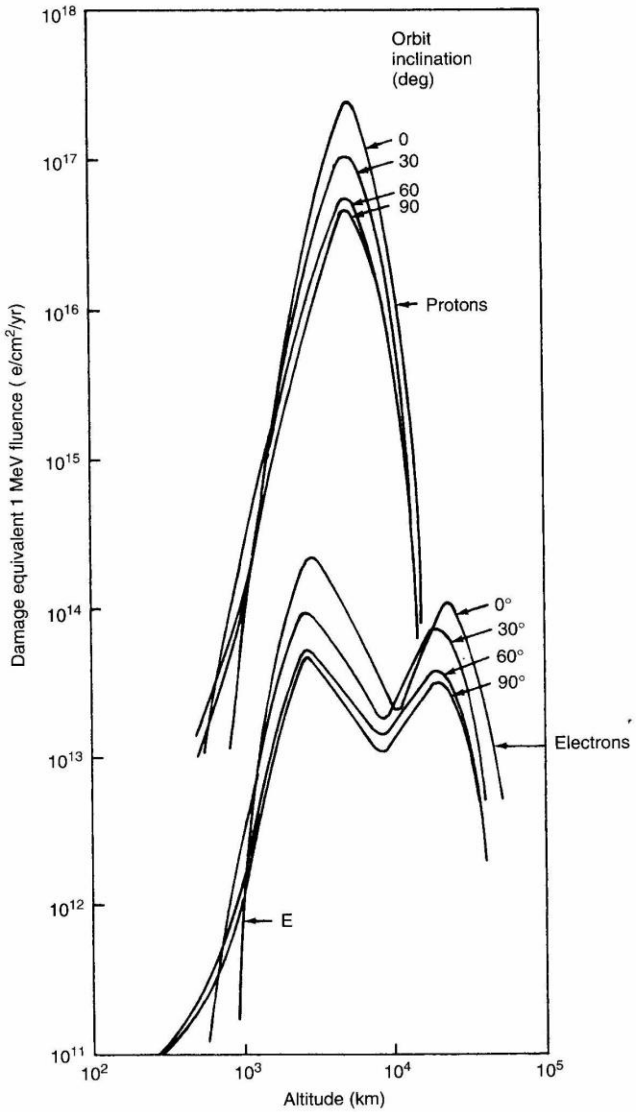


Figure 10.7 Damage equivalent 1 MeV fluence caused by electrons and protons due to trapped particles, to silicon cells protected by 150 μm fused silica covers and infinitely thick rear shielding [5] (From *Solar cell array design handbook* by Rauschenbach, H. S.

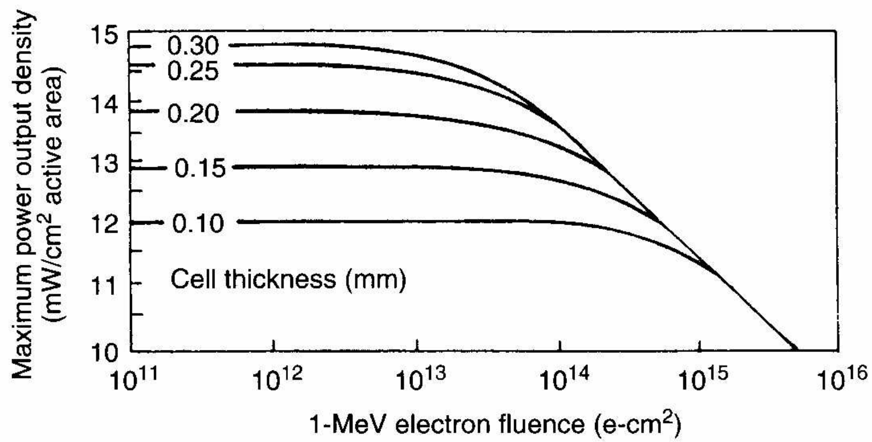


Figure 10.8 Effects of thickness and fluence on conventional non-p^+ silicon solar cell performance [5] (From *Solar Cell Array Design Handbook* by Rauschenbach, H. S.

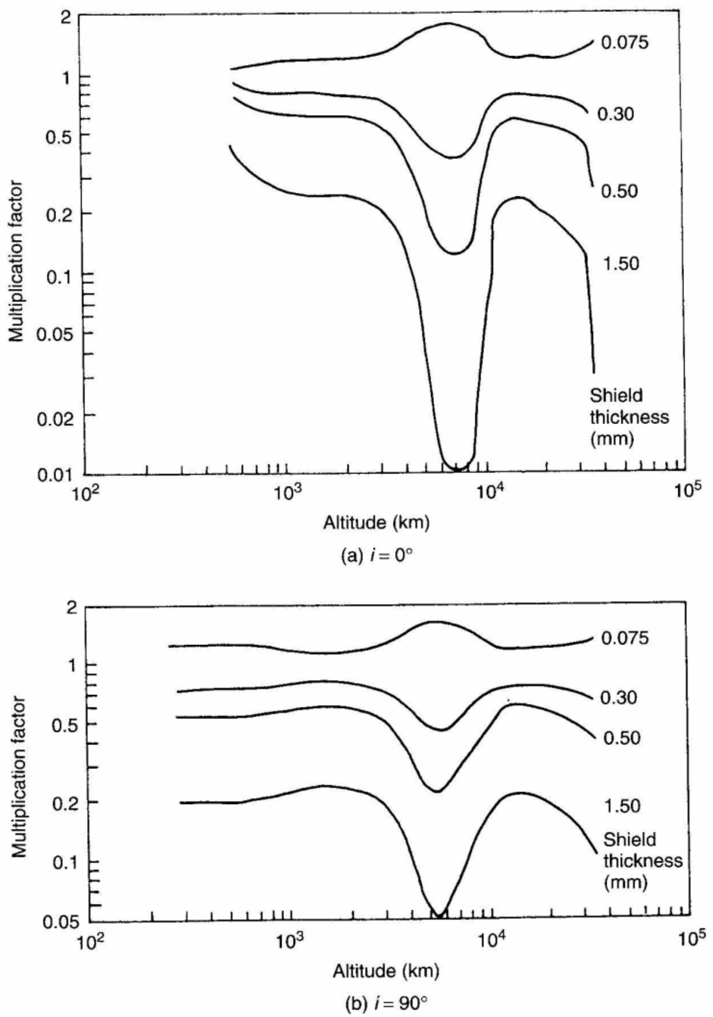


Figure 10.9 Multiplicative factors to be applied to damage fluence on a solar cell as a function of cover slip (shield) thickness, and operational orbit height (From *Solar Array Design Handbook* by Rauschenbach, H. S. Copyright © 1980 Van Nostrand Reinhold. All rights reserved)

Table 10.2 Solar array performance figures

Array	Type	BOL power (kW)	Specific mass (W/kg)	Power density (W/m^2)
XMM	Rigid	2.5	32	215
Astra 2B	Rigid	9.2	52	409
Comets	Flexible	6.3	34	146

Note: XMM: X-ray multi-mirror mission.

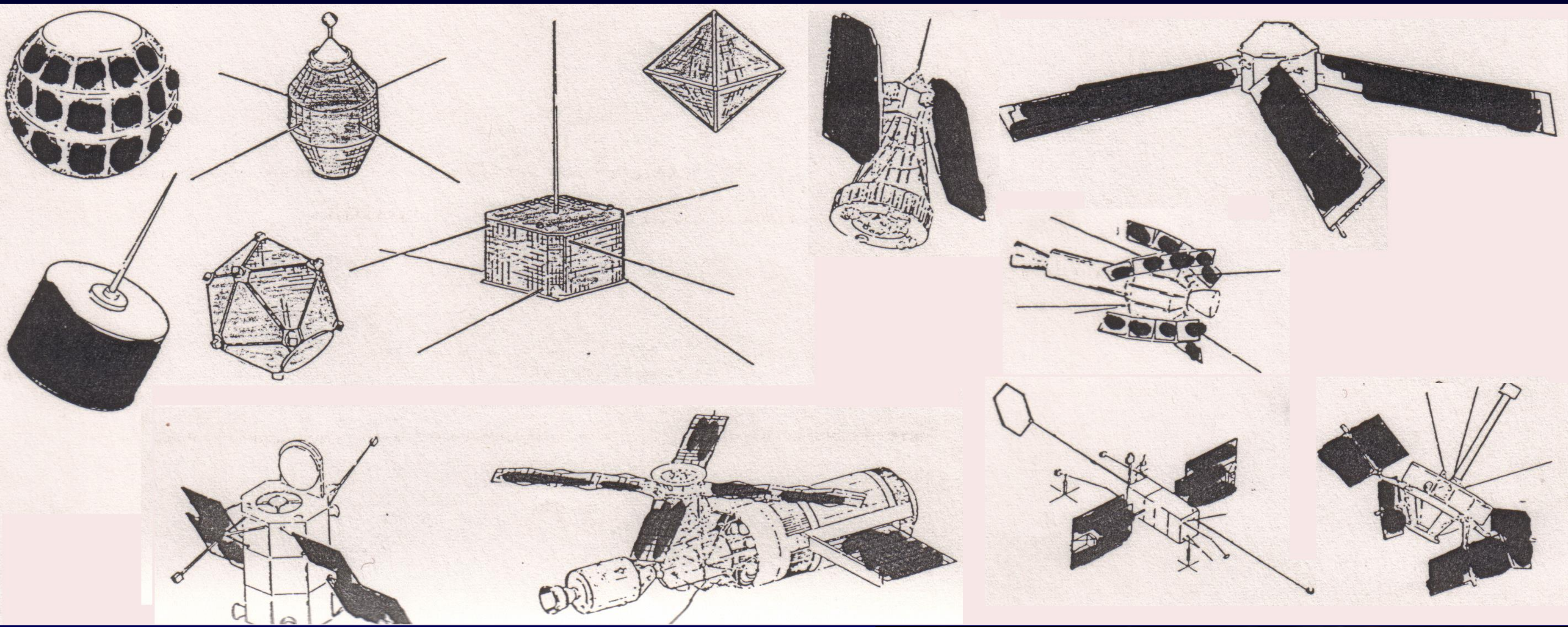
Area of Solar Cell for a 5 year EOL

- Using figures 10.7 to 10.9 it is possible to derive the area of active solar cells required to meet a specific mission requirement for end of life (EOL) performance.
- Suppose an output of 1kW is required at EOL for a satellite in a circular, equatorial orbit at 1000 km
- Silicon cells, 100 microns thick having the properties given in figure 10.8. Mission duration is 5 years
- From figure 10.7 find the total damage equivalent 1MeV electron fluence for a cell protected by a 150 micron cover slip.
- At 1000km damage equivalent due to protons is 1.7×10^{14} electrons/cm^2 /year and that due to electrons is 2×10^{12} electrons/cm^2/year.
- Total flux in 5 years is therefore $5(1.7 \times 10^{14})$ electrons/cm^2/year or 8.6×10^{14} electrons/cm^2/year
- From figure 10.8 the power per unit area is 11.5 mW/cm^2
- Hence: active area = 1000/ 0.0115 cm^2 or 8.7 m^2
- Using Triple Junction Technology the area can be reduced by 60%, hence 3.5 m^2

Solar Arrays

- Solar arrays using Silicon cells are made of cells that are generally 2cm x 4cm, having a conversion efficiency of 12 ~14%. Triple junction cells will give efficiencies of 40%
- Interconnections between cells represent a major array failure hazard. This arises because of the thermal cycling inherent upon entry/departure from sunlight to eclipse. Thermal stress relieving loops are required to reduce such mechanisms as interconnect lift-off and fracture.
- Silver interconnects are frequently used but are oxidised by atomic oxygen. Molybdenum provides better interconnect performance.
- A variety of substrate materials have been used and proven in space. Frequently Kapton with glass or carbon-fibre reinforcement, ~100 micron thick, forms the immediate interface with the cell, which may be mounted on a honeycomb panel for rigidity (Tracking and data relay satellite (TDRS) solar array)

Example Configurations of Solar cells on Spacecraft:



International Space Station

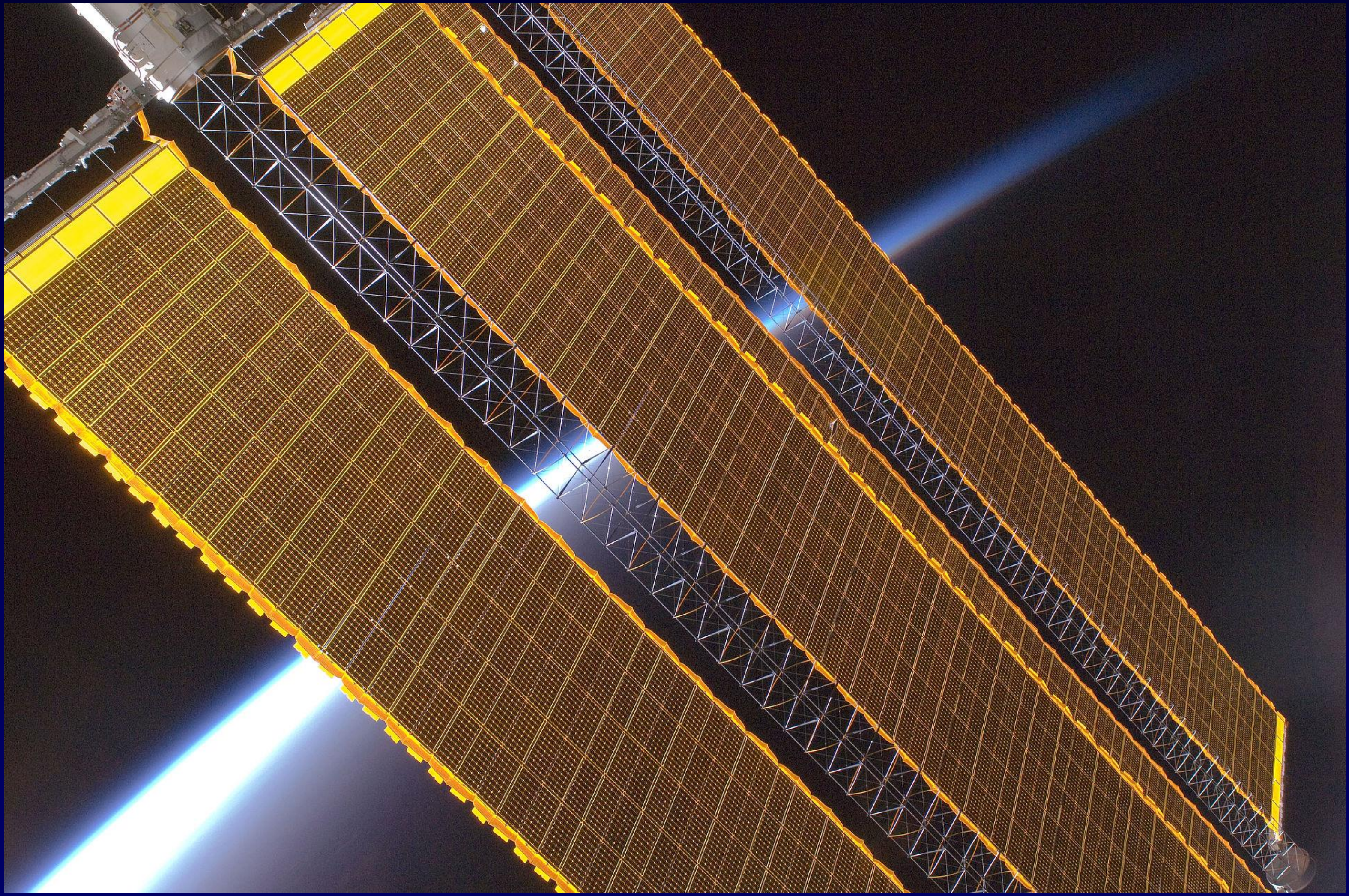
Typical Solar Cell Mounting

- **Body Mounted**
  Ideal for spinning spacecraft, easy temperature control, instantaneous power depends on projected area, cylindrical s/c best but limited orientations with respect to sun

- **Deployable Arrays/Paddles**
  Structure types: rigid; semi-rigid; flexible
  Temperature extremes
  Orientation with respect to spacecraft:
  (a) fixed - limits spacecraft orientation
  (b) controlled orientation – always perpendicular to sun
  most efficient use of cells

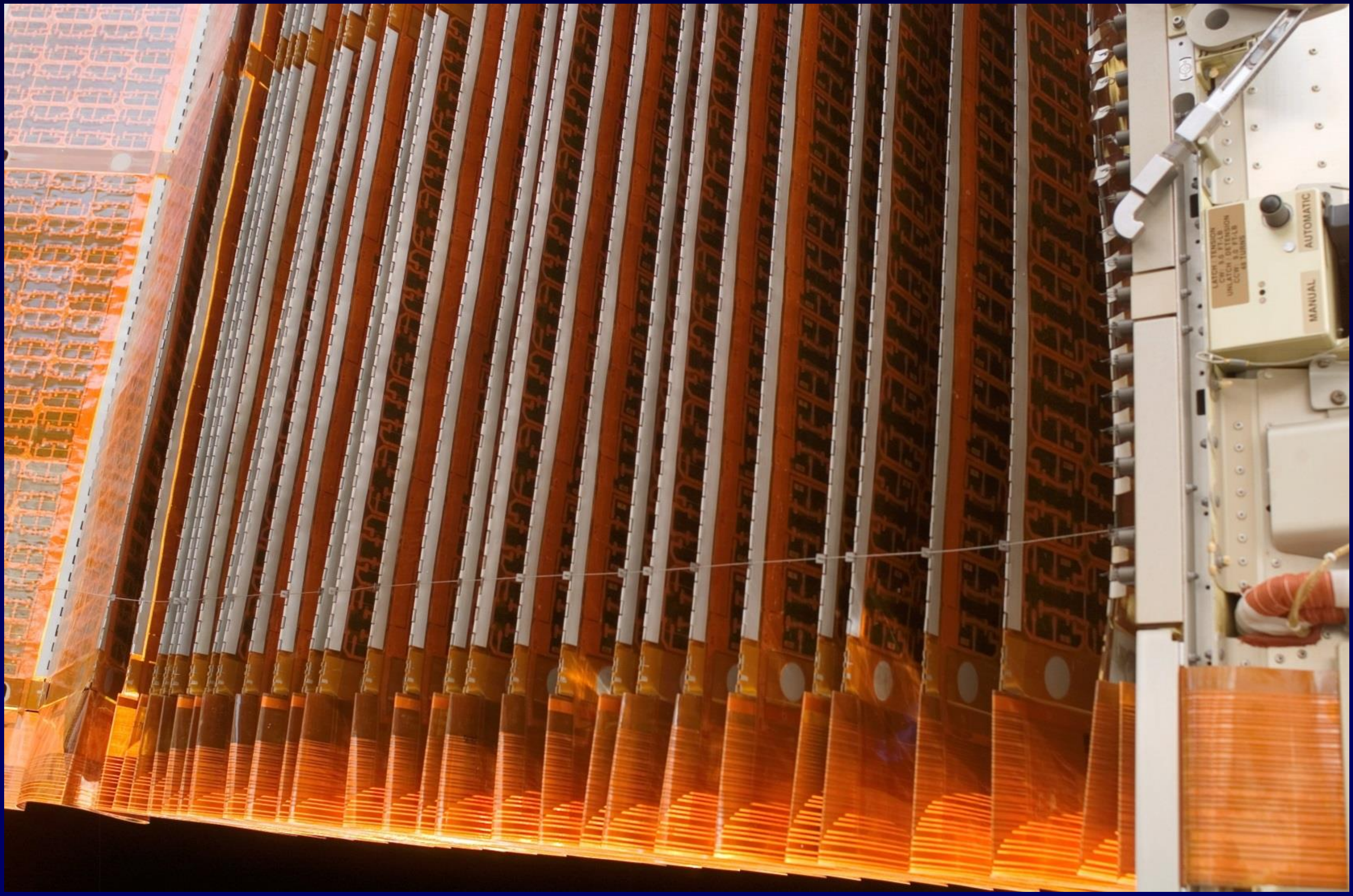
- **Design Considerations**
  a) Launch vibration
  b) Exposed to thermal stress – cracks wires, cell substrate
  -170° to $+60^{\circ}$ at geostationary
  c) Power output degrades with time
  radiation effects semiconductor
  micrometeorites reduce transparency of window

ISS solar array wing - 8 wings - each 420 m^2 – giving 3360 m - 120 kW



Folded solar array - ISS

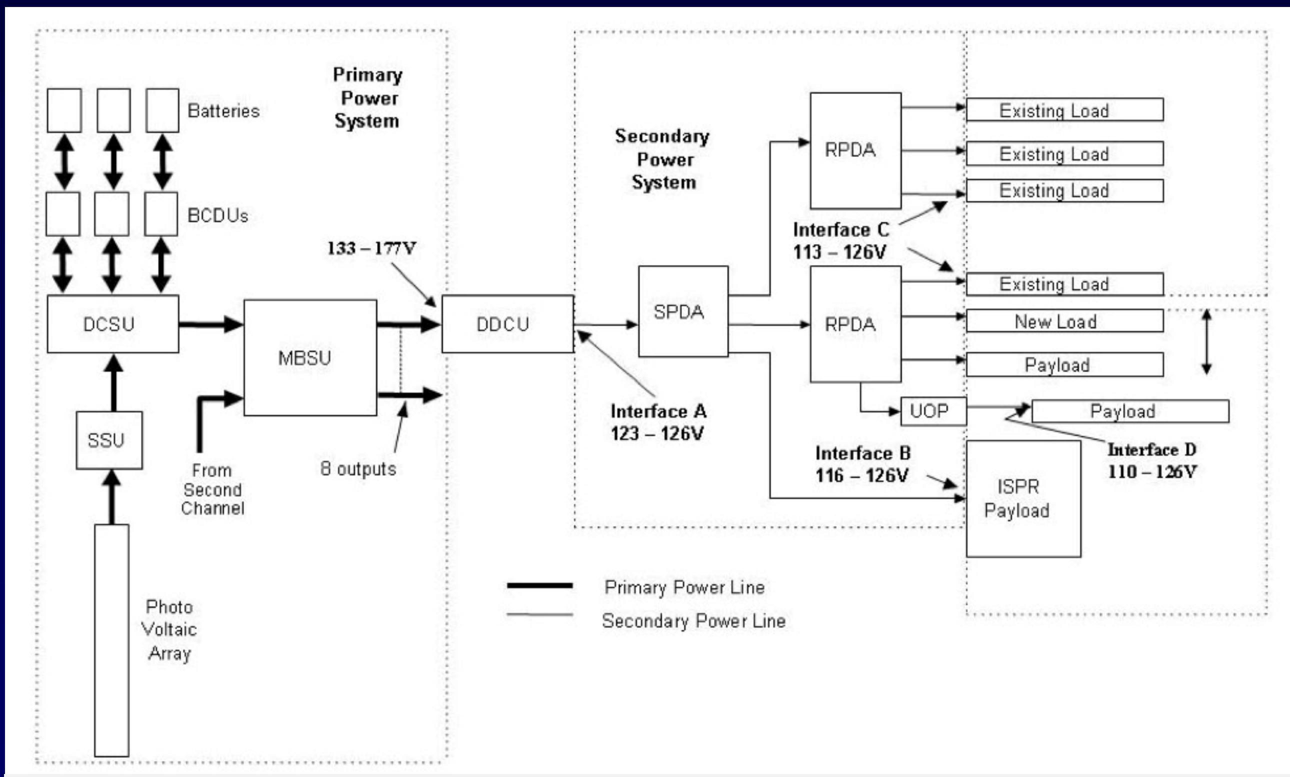
Retractable "blankets" of solar cells with a mast between them. Each wing uses nearly 33,000 solar cells and when fully extended is 35 metres in length and 12 metres wide. When retracted, each wing folds into a solar array blanket box just 51 centimetres high and 4.57 metres in length.



ISS Electrical Power Schematic - BCDU Battery Charge Discharge Unit, DCSU DC Switching Unit, SSU Sequential Shunt Unit, MBSU Main Bus Switching Unit, DDCU DC/DC Converter Unit

Nickel-hydrogen batteries to provide continuous power during the "eclipse" part of the orbit; 35 minutes of every 90 minute orbit

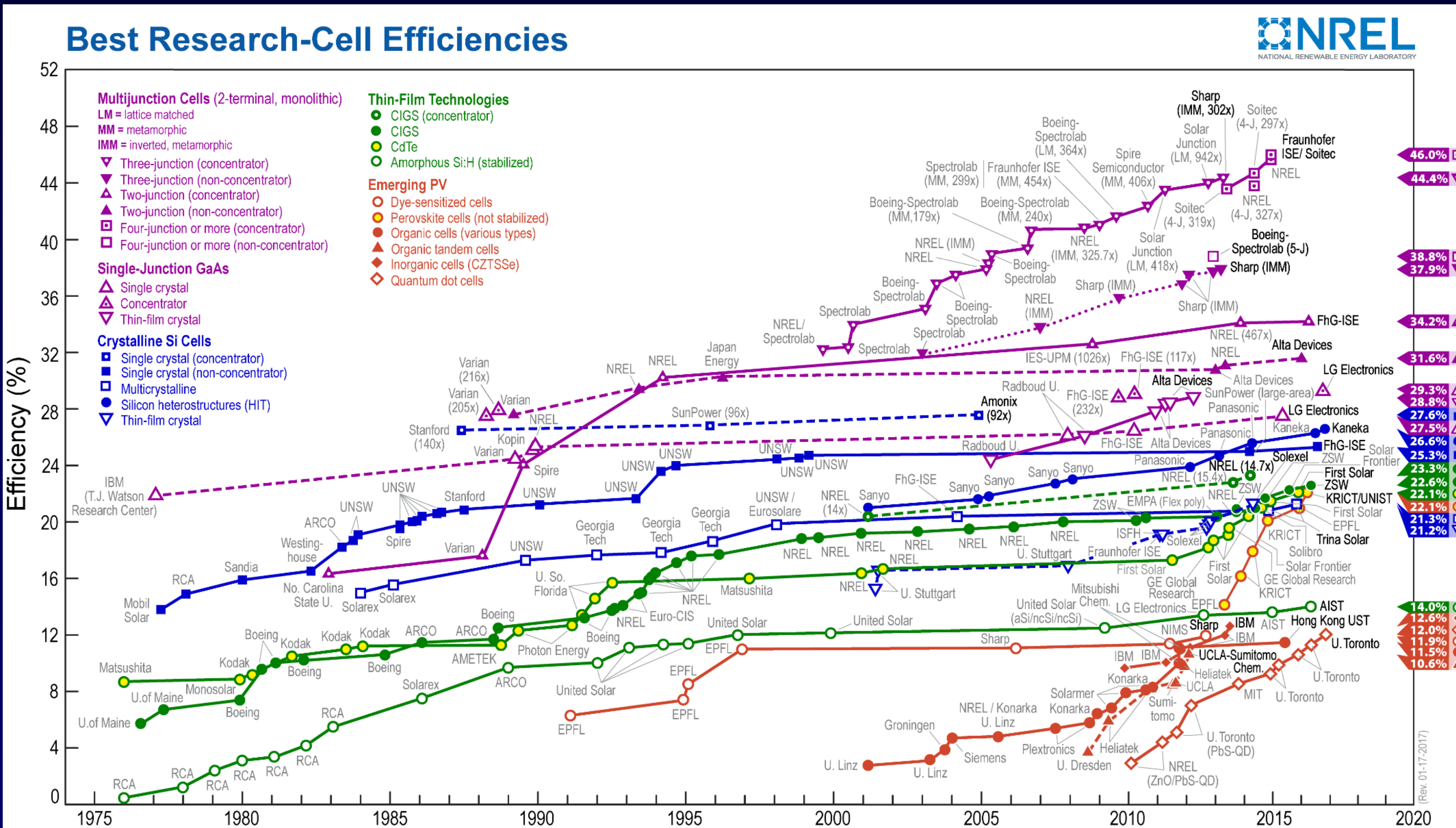
Battery charge/discharge units (BCDUs)



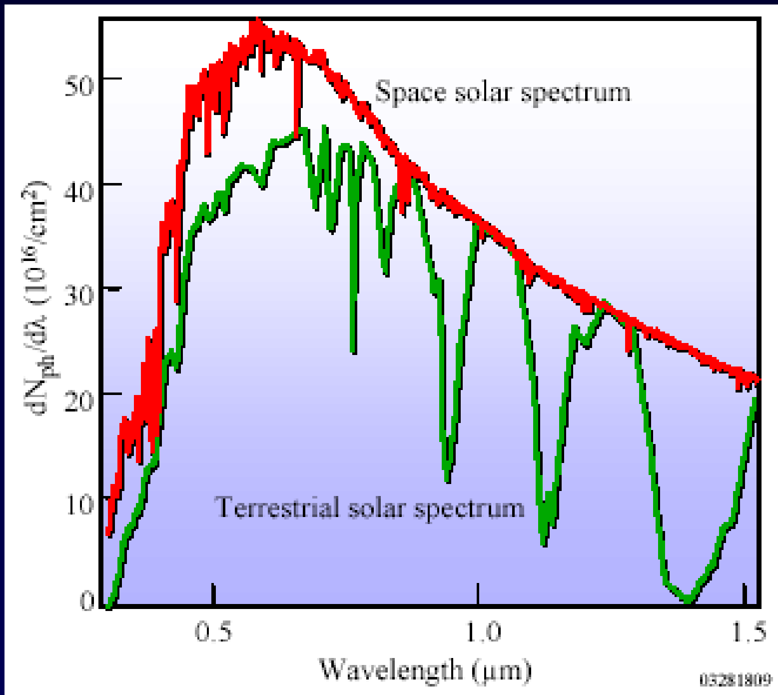
The power management and distribution subsystem operates at a primary bus voltage set to V_{mp}, the peak power point of the solar arrays. As of December 30, 2005, V_{mp} was 160 volts DC.

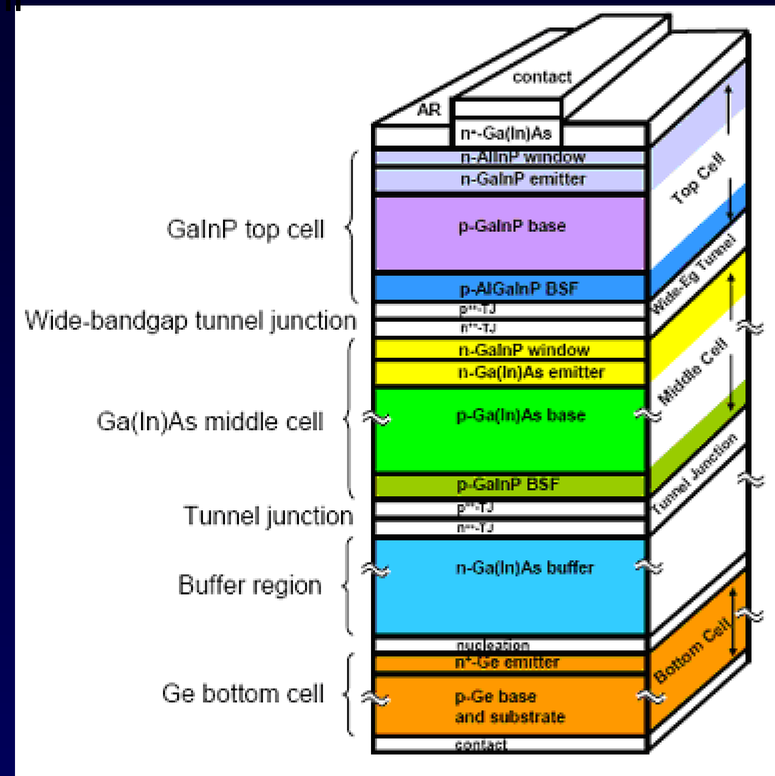
Eighty-two separate solar array strings feed a sequential shunt unit (SSU) that provides coarse voltage regulation at the desired V_{mp}. The SSU applies a "dummy" (resistive) load that increases as the station's load decreases (and vice versa) so the array operates at a constant voltage and load.

Future Efficiency of Solar Cells – Disruptive Technology



Structure of a triple-junction photovoltaic cell



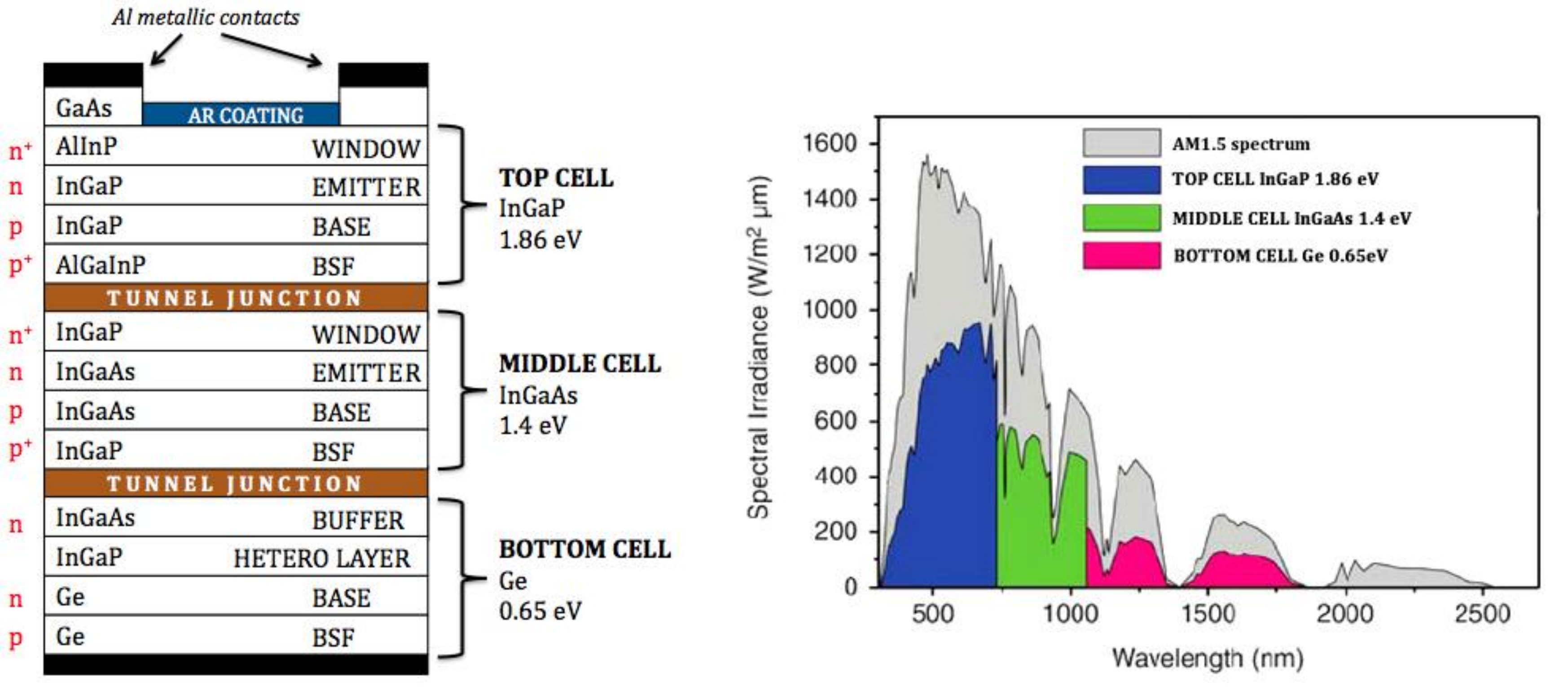


The highest-efficiency solar cells use multiple materials with bandgaps that span the solar spectrum. Multi-junction solar cells consist of some single-junction solar cells stacked upon each other, so that each layer going from the top to the bottom has a smaller bandgap than the previous, and so it absorbs and converts the photons that have energies greater than the bandgap of that layer and less than the bandgap of the higher layer

^ a b c d N.V.Yastrebova (2007). *High-efficiency multi-junction solar cells: current status and future potential*. http://sunlab.site.uottawa.ca/pdf/whitepapers/HiEfficMjSc-CurrStatus&FuturePotential.pdf.

(a) **The structure of a MJ solar cell. There are six important types of layers: pn junctions, back surface field (BSF) layers, window layers, tunnel junctions, anti-reflective coating and metallic contacts.**

(b) Graph of spectral irradiance E vs. wavelength λ over the AM 1.5 solar spectrum



(a) (b)

Triple Junction Photovoltaics

- The choice of materials for each sub-cell is determined by the requirements for lattice-matching, current-matching, and high performance opto-electronic properties.

- each sub-cell is connected electrically in series, the same current flows through each junction. The materials are ordered with decreasing bandgaps, E_g, allowing sub-bandgap light ($hc/\lambda < e \cdot E_g$) to transmit to the lower sub-cells.

- Therefore, suitable bandgaps must be chosen such that the design spectrum will balance the current generation in each of the sub-cells, achieving current matching.

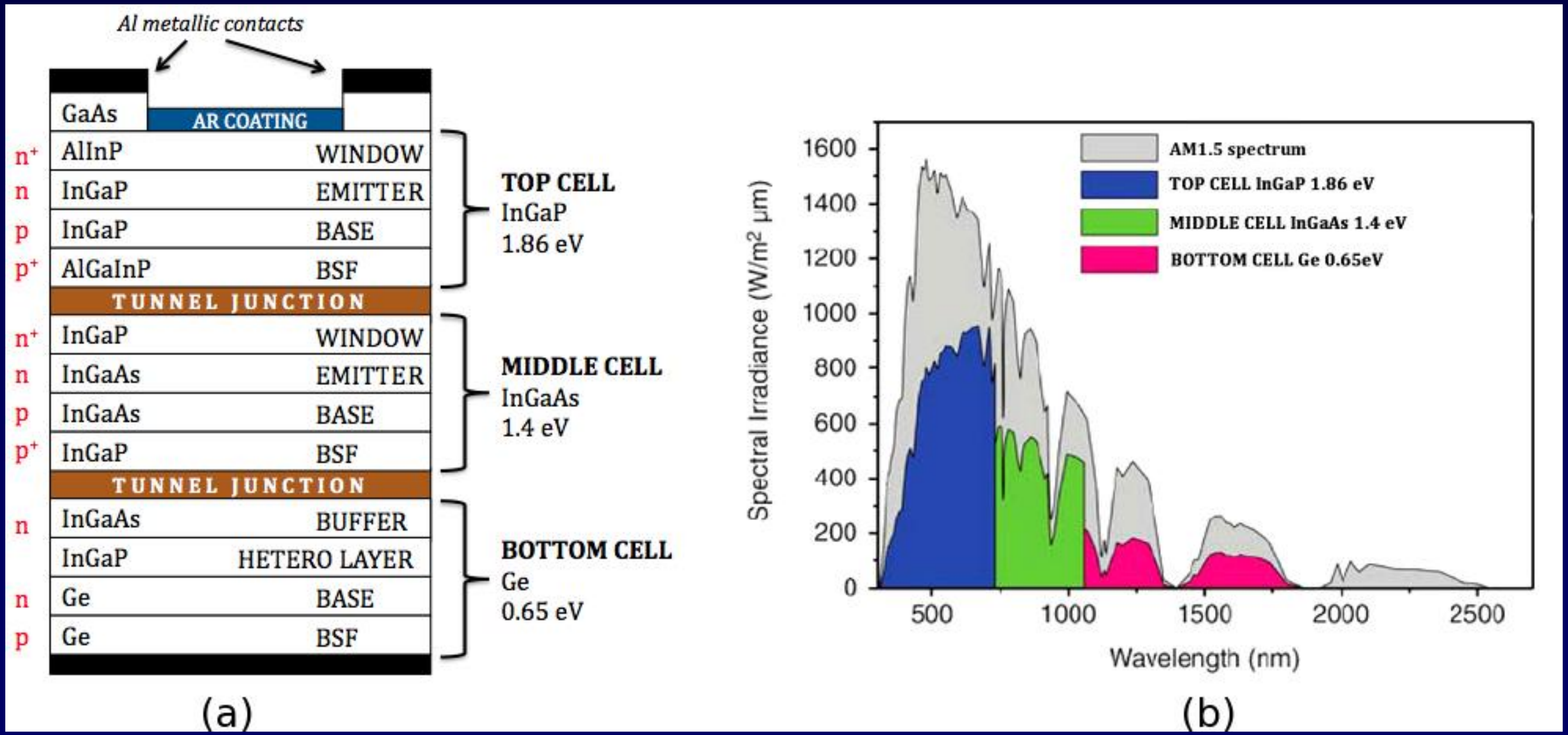
The favorable values in the table below justify the choice of materials typically used for multi-junction solar cells: InGaP for the top sub-cell (E_g = 1.8 - 1.9 eV), InGaAs for the middle sub-cell (E_g = 1.4 eV), and Germanium for the bottom sub-cell (E_g = 0.67 eV). The use of Ge is mainly due to its lattice constant, robustness, low cost, abundance, and ease of production.

Material	E_g, eV Bandgap energy	*a*, nm Crystal lattice constants	Absorption (λ = 0.8 μm), 1/μm	μ_n, cm²/(V·s) Mobility	τ_p, μs Life time	Hardness (Mohs)	α, μm/K Absorption coefficient	*S*, m/s Surface recombination velocity
c-Si	1.12	0.5431	0.102	1400	1	7	2.6	0.1–60
InGaP	1.86	0.5451	2	500	–	5	5.3	50
GaAs	1.4	0.5653	0.9	8500	3	4–5	6	50
Ge	0.65	0.5657	3	3900	1000	6	7	1000
InGaAs	1.2	0.5868	30	1200	–	–	5.66	100–1000

- Because the different layers are closely lattice-matched, the fabrication of the device typically employs metal-organic chemical vapor deposition (MOCVD).
- This technique is preferable to the molecular beam epitaxy (MBE) because it ensures high crystal quality and large scale production.

Material Properties

- The layers must be electrically optimal for high performance. This necessitates usage of materials with strong absorption coefficients $\alpha(\lambda)$, high minority carrier lifetimes $\tau_{minority}$, and high mobilities μ.



Tunnel Junction

The main goal of tunnel junctions is to provide a low electrical resistance and optically low-loss connection between two subcells.

Without it, the p-doped region of the top cell would be directly connected with the n-doped region of the middle cell.

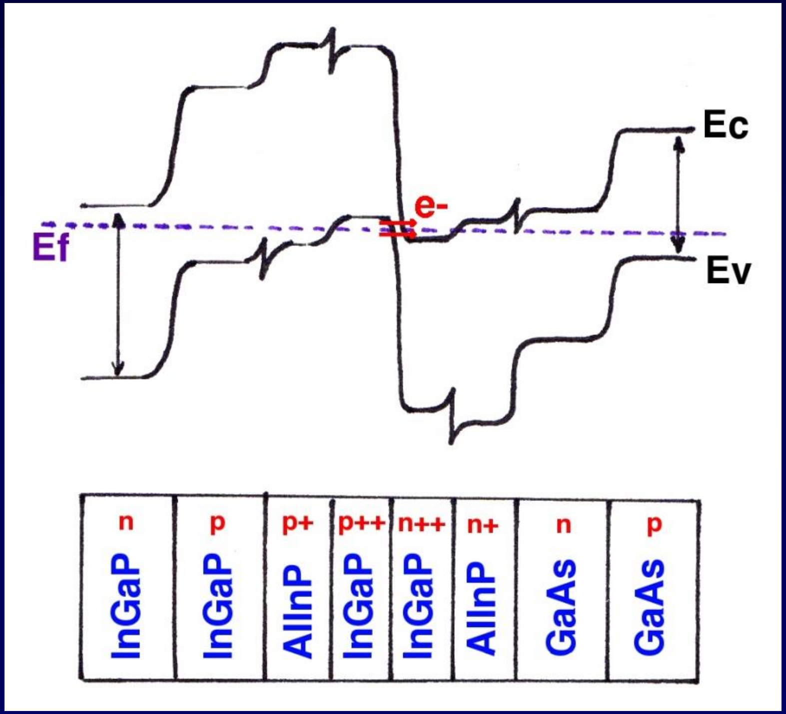
Hence, a pn junction with opposite direction to the others would appear between the top cell and the middle cell.

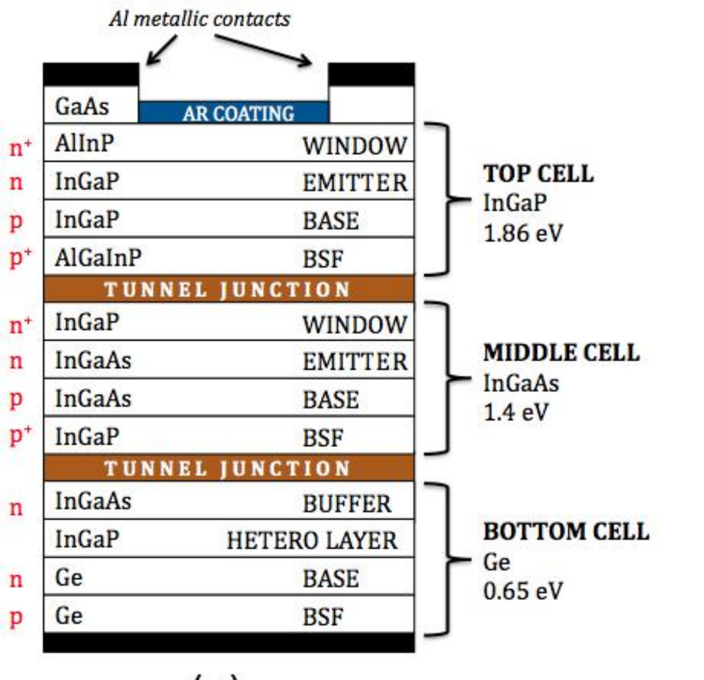
Consequently, the photovoltage would be lower than if there would be no parasitic diode.

In order to decrease this effect, a tunnel junction is used. It is simply a wide band gap, highly doped diode.

The high doping reduces the length of the depletion region. Hence, electrons can easily tunnel through the depletion region.

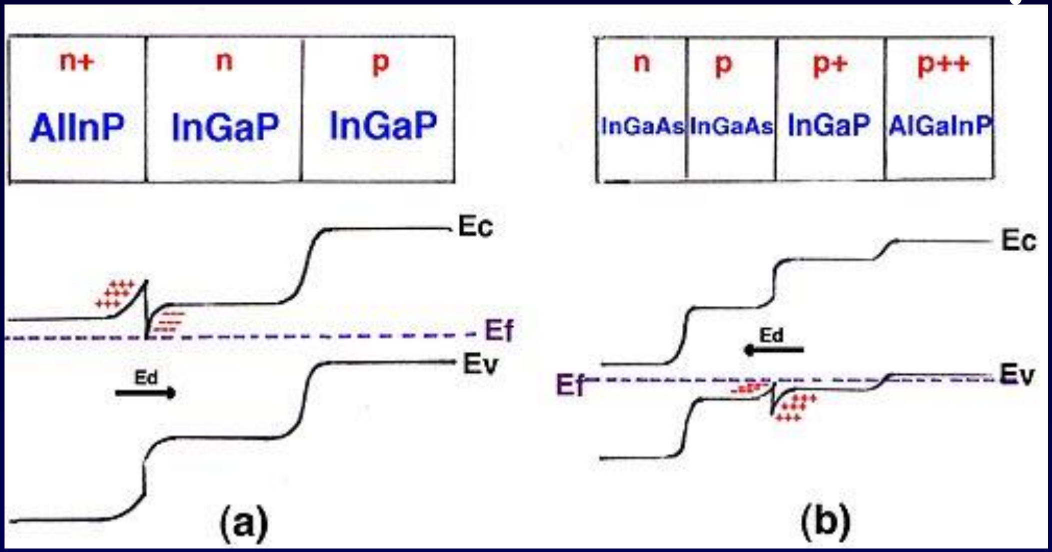
Layers and band diagram of the tunnel junction. Because the length of the depletion region is narrow and the band gap is high, electrons can tunnel.





(a)

Window layer and back-surface field



- (a) Layers and band diagram of a window layer. The surface recombination is reduced.
- (b) Layers and band diagram of a BSF layer. The scattering of carriers is reduced.

- A window layer is used in order to reduce the surface recombination velocity *S*.

Similarly, a back-surface field (BSF) layer reduces the scattering of carriers towards the tunnel junction. The structure of these two layers is the same: it is a heterojunction which catches electrons (holes). Indeed, despite the electric field E_d, these cannot jump above the barrier formed by the heterojunction because they don't have enough energy, as illustrated in the figure. Hence, electrons (holes) cannot recombine with holes (electrons) and cannot diffuse through the barrier.

Boeing 702MP Satellite

- The payload is powered by a solar array consisting of ultra triple-junction gallium arsenide solar cells. Intelsat have 9 of these

Solar Panels	13.6 to 18 kw, beginning of life
Construction	Two wings each with three to four panels of Ultra Triple-Junction (UTJ) gallium arsenide solar cells
Batteries	24 to 40 cell Li-Ion, 236 Ahr

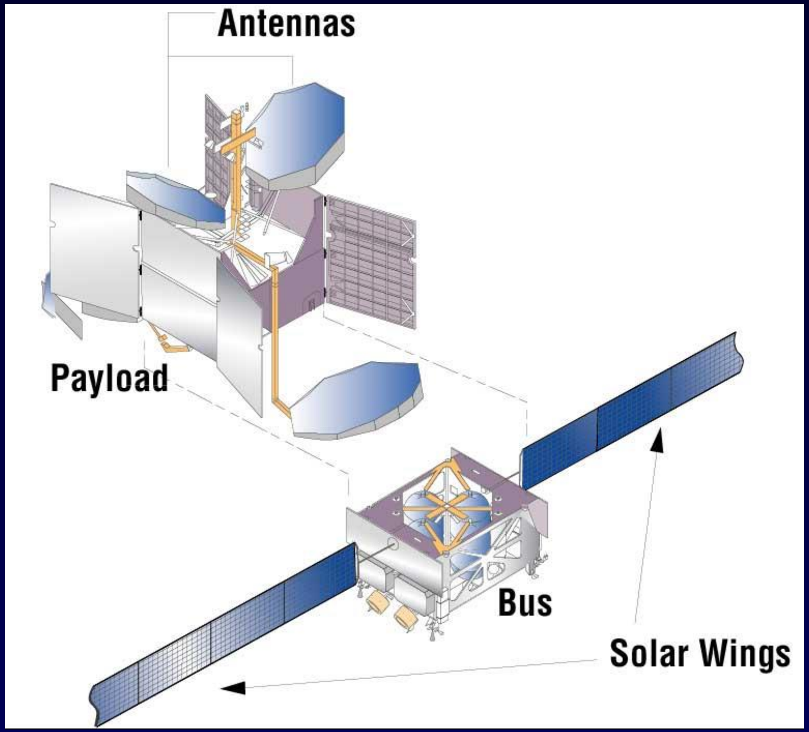
In Orbit	H, 5.8 m to 8.6 m W, antennas: 9.2 m L, solar arrays: 36.9 m to 38.1 m
Stowed	H, 5.8 m W, 3.6 m x 3.1 m
Mass at Launch	5,800 kg to 6,160 kg 80-w TWTAs or hosted payload
Mass in Orbit (beginning of life)	3,582 kg to 3,833 kg

Boeing 702HP – Inmarsat 5 Comms satelite

- Each Inmarsat-5 satellite carries 89 Ka-band beams that operate in geosynchronous orbit with flexible global coverage. The satellites are designed to generate approximately 15 kilowatts of power at the start of service and approximately 13.8 kilowatts at the end of their 15-year design life. Each spacecraft's two solar wings employ five panels each of ultra triple-junction gallium arsenide solar cells.

- The Inmarsat-5 satellites will provide Inmarsat with a comprehensive range of global mobile satellite services, including:
- mobile broadband communications for deep-sea vessels, in-flight connectivity for airline passengers and streaming high-resolution video, voice and data.
- The Boeing 702HP carries the xenon ion propulsion system (XIPS) for all on-orbit manoeuvring.

Boeing 702HP Satellite

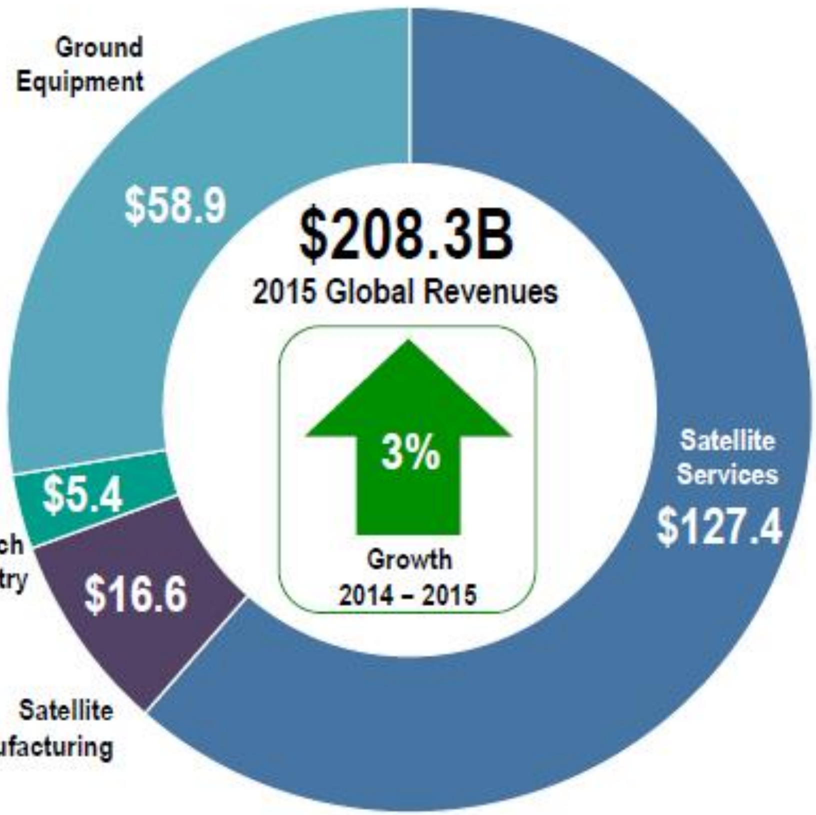


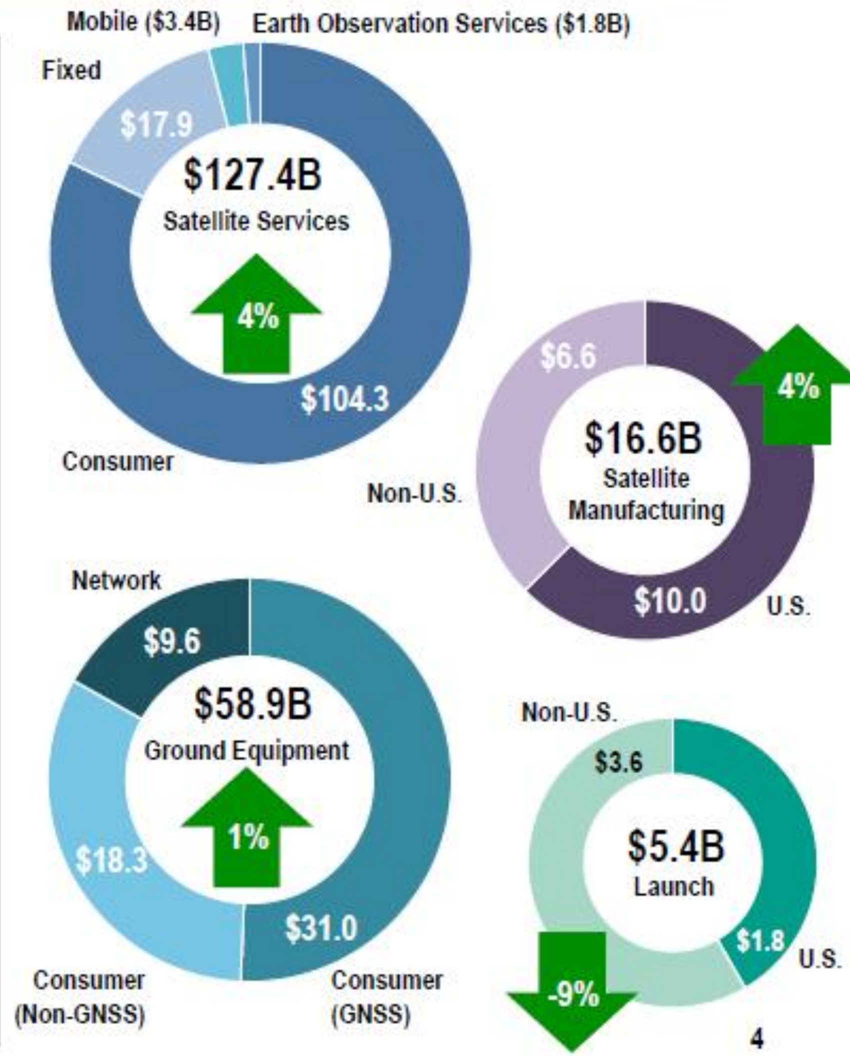
- Innovation extends to the Boeing 702HP power systems as well. The Boeing 702 offers a range of power up to 18 kW. Dual and triple-junction gallium arsenide solar cells enable such high power levels.
- The 702's has advanced xenon ion propulsion system (XIPS), which was pioneered by Boeing. XIPS is 10 times more efficient than conventional liquid fuel systems.
- Four 25-cm thrusters provide economical stationkeeping, needing only 5 kg of fuel per year - a fraction of what bipropellant or arcjet systems consume. Using XIPS for final orbit insertion conserves even more mass as compared to using an on-board liquid apogee engine.

http://exnetapps.intelsat.com/flash/coverage-maps/index.html

- Use the website above to see what Intelsat satellites are available

2015 Satellite Industry Indicators Summary





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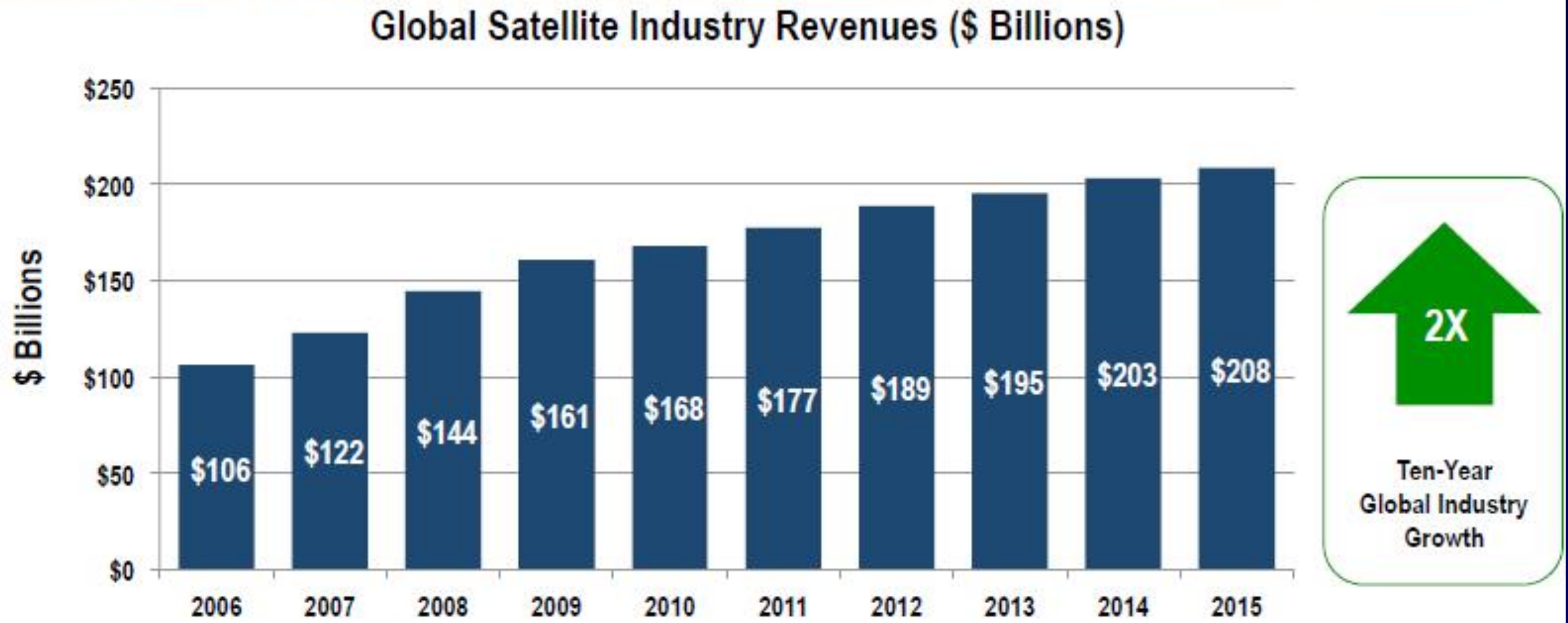
Global Satellite Industry Revenues



Growth Rate	19%	15%	18%	11%	5%	6%	7%	3%	4%	3%

Global satellite industry grew 3% in 2015, slightly above worldwide economic growth (2.4%) and U.S. growth (2.5%)

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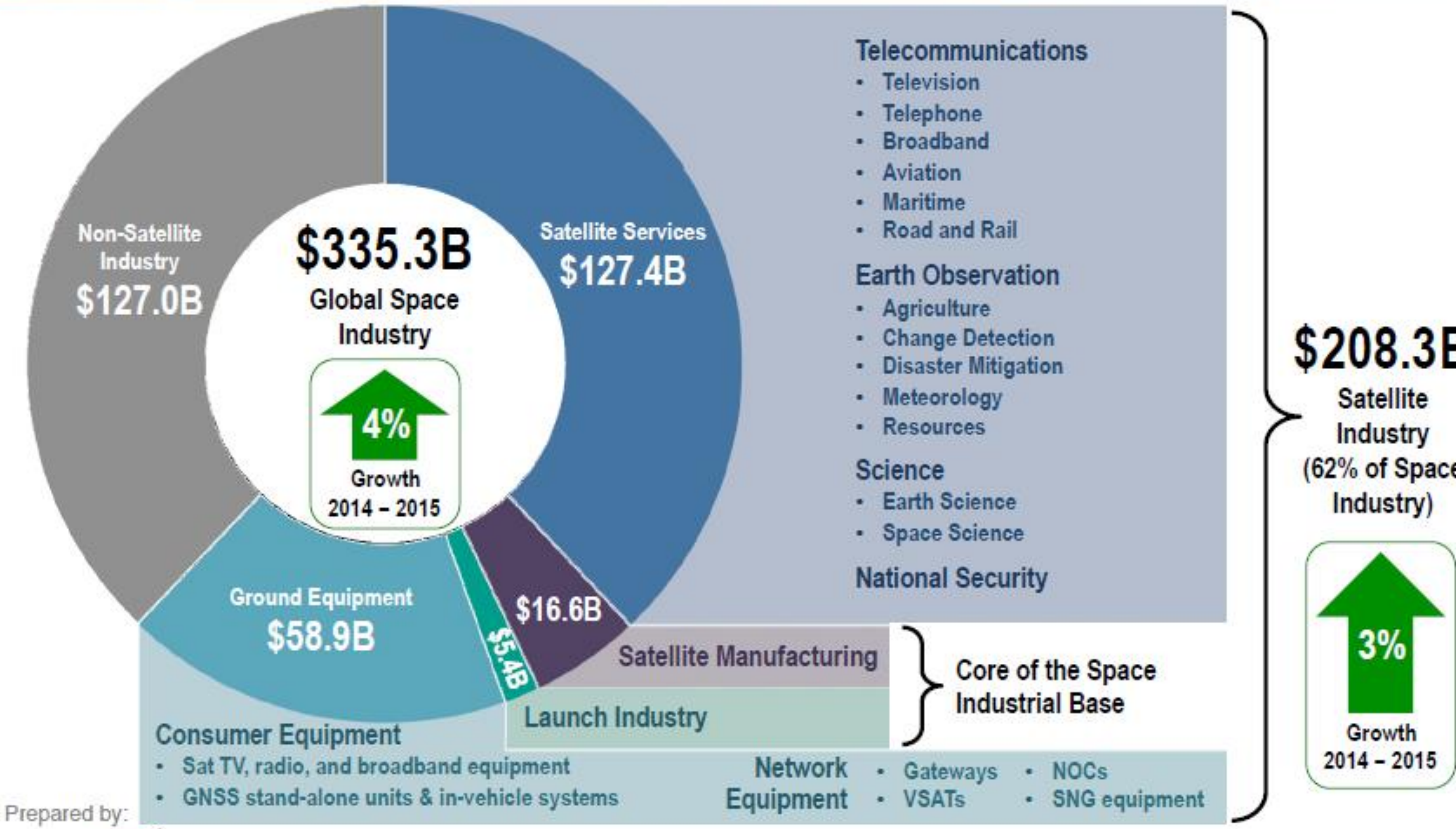
The Satellite Industry in Context



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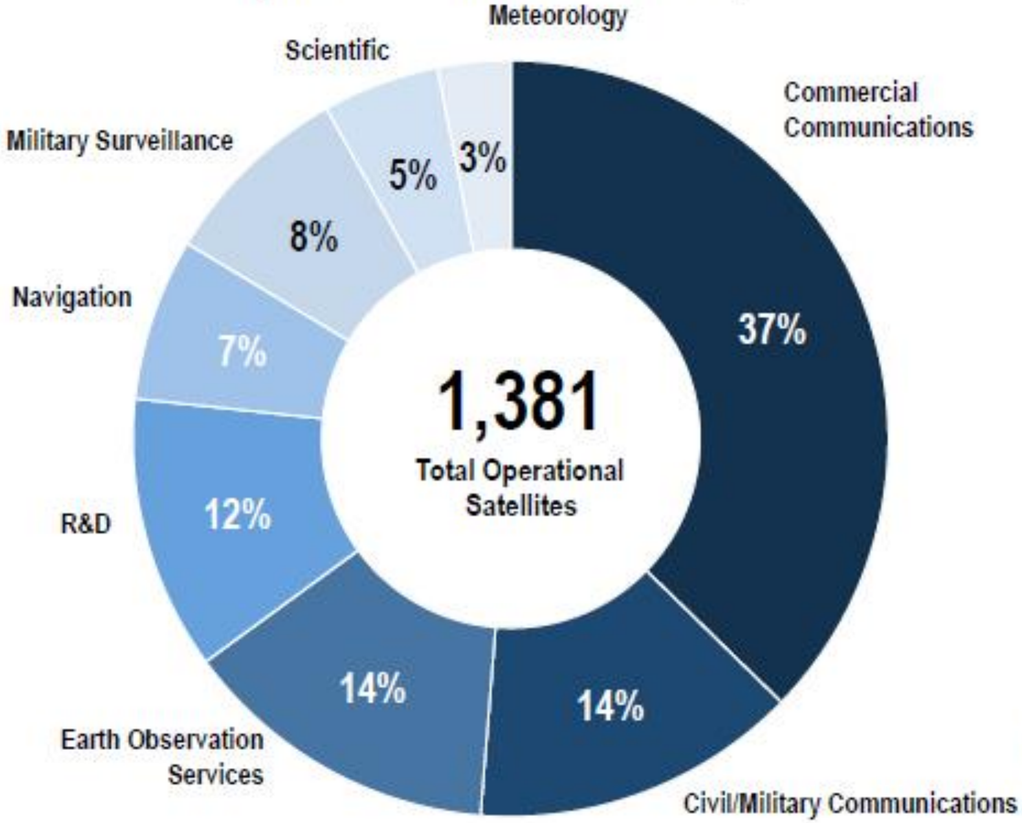
Notes: Network operations centers (NOCs), satellite news gathering (SNG), very small aperture terminal (VSAT) equipment, global navigation satellite systems (GNSS)

The Satellite Network in Context

Operational Satellites by Function

(as of December 31, 2015)



- Number of satellites increased 39% over 5 years, compared to 986 reported in 2011
  - » Average number of satellites launched per year in 2011-2015 increased 36% over previous 5 years
  - » Small and very small satellites deployed in LEO contribute to this growth
  - » Average operational lives of certain satellite types (such as GEO communications satellites) are becoming longer
- 59 countries with operators of at least one satellite (some as part of regional consortia)

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Radioisotope thermoelectric generators (RTG)

- For deep space missions, the use of fuel cells is precluded by their long duration
- Solar arrays produce less power as they move away from the Sun
- For spacecraft travelling further than Mars, solar arrays show disadvantages from a system viewpoint, compared with radioisotope generators (Triple Junction technology has changed this conclusion, which is now in debate)
- The operation of an RTG is based on the thermoelecric effect noted by Seebeck, that it is possible to generate a voltage between two terminals, A and B (either conductors or semiconductors) if a temperature difference is maintained
- Practical RTG systems utilise two semiconductor materials – one *p-type*, the other *n-type, see figure*

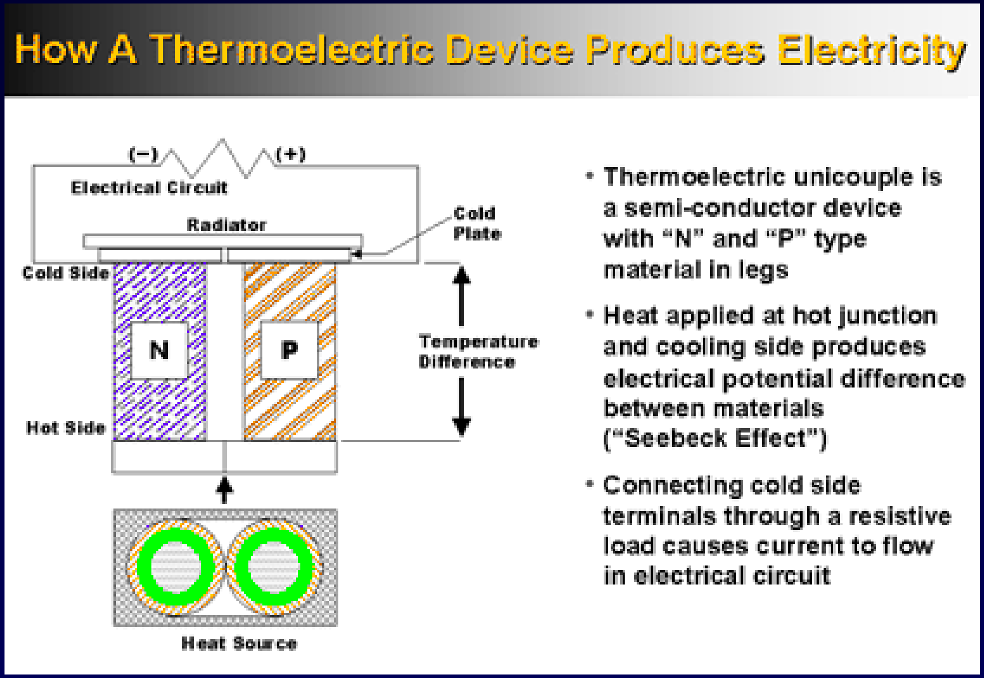
RTG Successes

- Cassini's electrical power source - radioisotope thermoelectric generators (RTGs) - have provided electrical power for some of the U.S. space program's greatest successes
- Including the Apollo lunar landings and the Viking Landers that searched for life on Mars.
- RTGs made possible National Aeronautics and Space Administration (NASA) celebrated Voyager explorations of Jupiter, Saturn, Uranus and Neptune, as well as the Pioneer missions to Jupiter and Saturn.
- RTG power sources enabled the Galileo mission to Jupiter and the international Ulysses mission studying the Sun's polar regions.

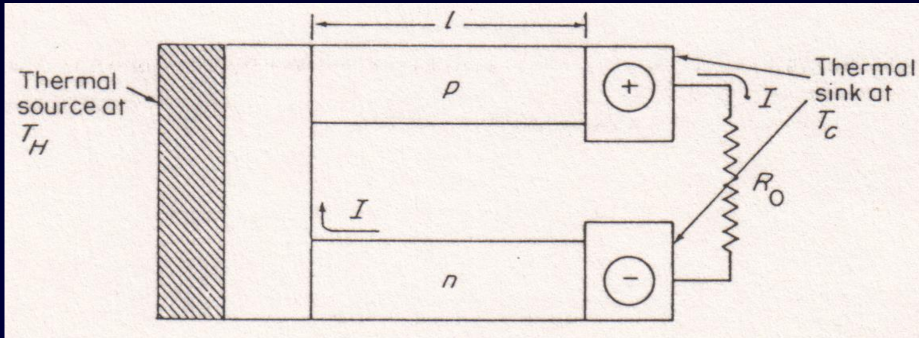
Radioisotope thermoelectric generators (RTG)

- The power output from such a device is a function of the absolute temperature of the hot junction, the temperature difference that may be maintained between the junctions and also the properties of the materials
- Table 10.4 indicates that high specific power levels are available with shorter half lives and hence for shorter duration missions
- For deep space missions a long isotope life is essential
- For example, the design life for the Cassini- Saturn orbiter is 11 years, after which time the electrical power source is required to be 628 W. For these missions Plutonium is used exclusively.
- Systems for Nuclear Auxiliary Power (SNAP-19) which powered the Viking lander vehicle to Mars, had a specific power of 2.2 W/kg, with a thermal/electric efficiency of ~5%. The output electrical power was 35W

RTG's



RTGs Ready to be Shipped to Cape Canaveral



Radioisotope Thermal Generator, RTG

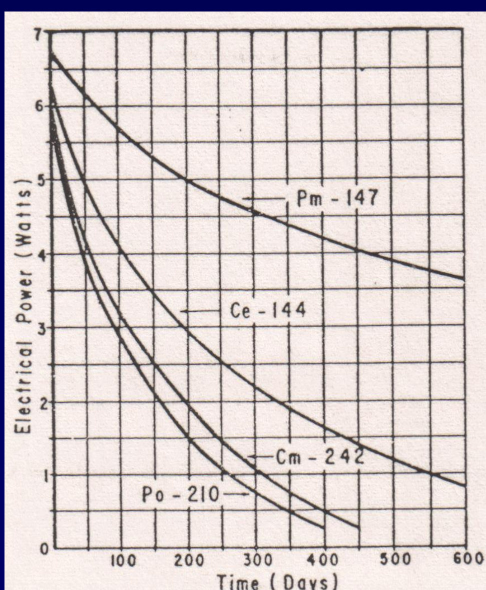
Only viable source for missions to the outer planets (beyond Mars) as solar intensity falls too low for high power solar cells.

Solar intensity percentage of 1 AU 1371± 5W/m²	
Mercury	667
Venus	191
Earth	100
Mars	43.1
Jupiter	3.69
Saturn	1.10
Uranus	0.27
Neptune	0.11
Pluto	0.064

Lump of 'hot' radioactive material provides heat source for thermo-electric generation via junction of dissimilar metals or semiconductors.

Source degrades with time. Exponential decay of electric power.

- **Need to be able to radiate away excess heat at early mission stage**
- **Need to mount on boom away from instruments sensitive to radiation**



For Jupiter, Saturn and beyond spacecraft use(d) RTGs

(Voyagers, Cassini, Pioneer, Ulysses, etc) Juno exception

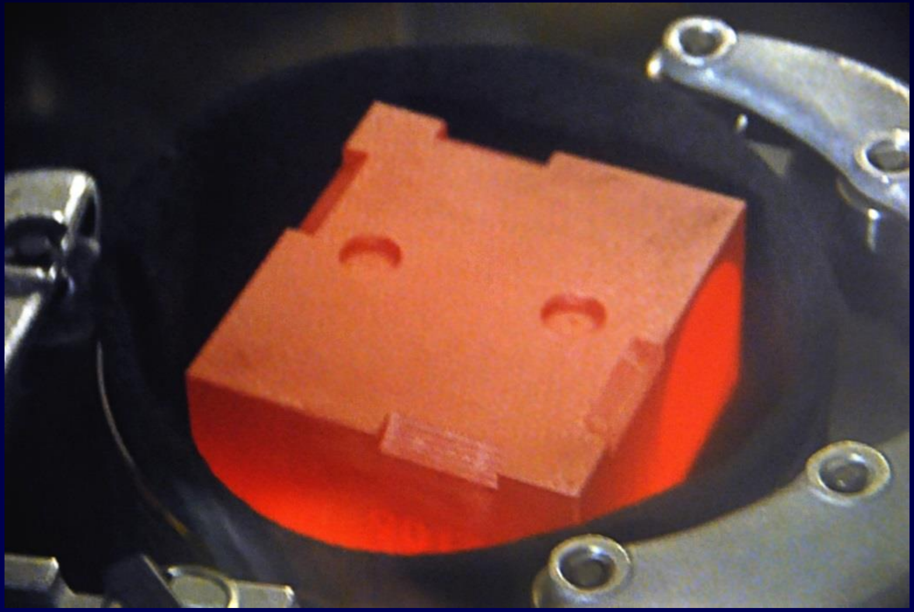
Environmental concern about possible launch failure or bad calculated trajectory on any swingbys past Earth

Isotope	Fuel form	Decay	Power density (W/g)	$\tau_{1/2}$(yr)
Polonium 210	GdPo	α	82	0.38
Plutonium 238	PuO_2	α	0.41	86.4
Curium 242	Cm_2O_3	α	98	0.4
Strontium 90	SrO	β	0.24	28.0

$$P_t = P_0 \exp\left(\frac{-0.693}{\tau_{1/2}} t\right)$$

Table 10.4

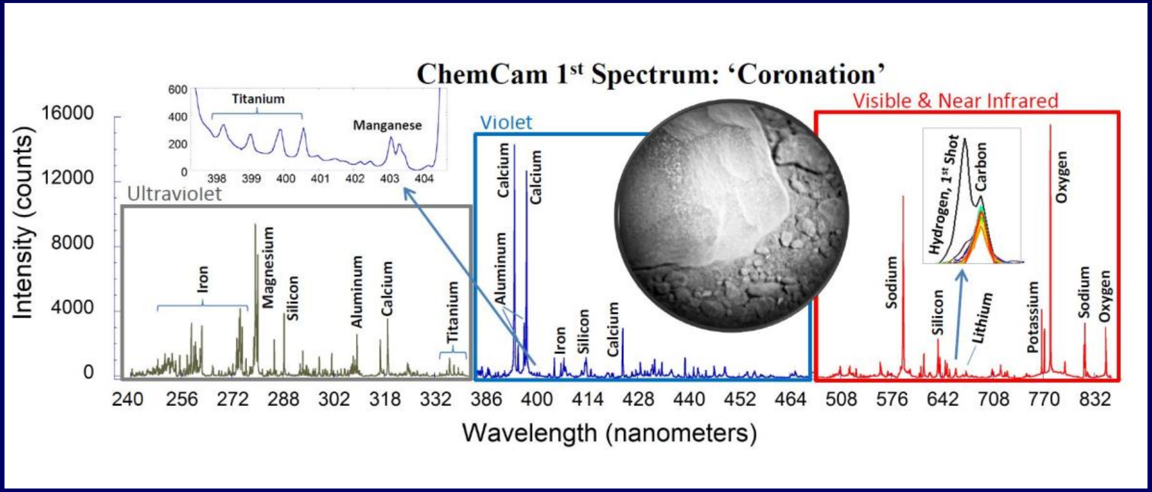
Plutonium-238 dioxide



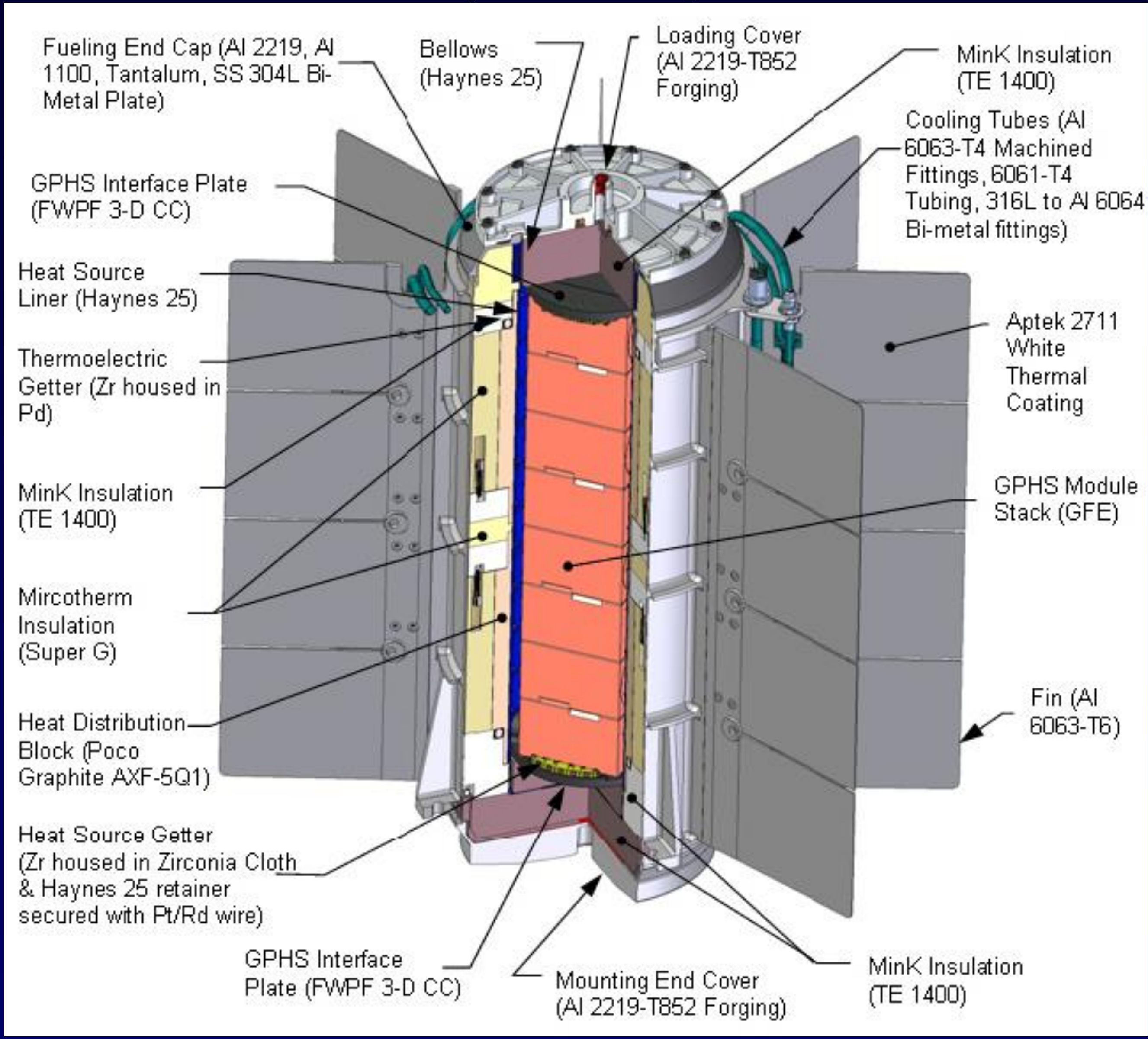
The Mars Science Laboratory's radioisotope power system was assembled by putting nuclear heat sources within graphite impact shells into high-strength carbon-carbon modules at Idaho National Laboratory.

- *Curiosity*'s RTG is fueled by 4.8 kg (11 lb) of plutonium-238 dioxide supplied by the U.S. Department of Energy

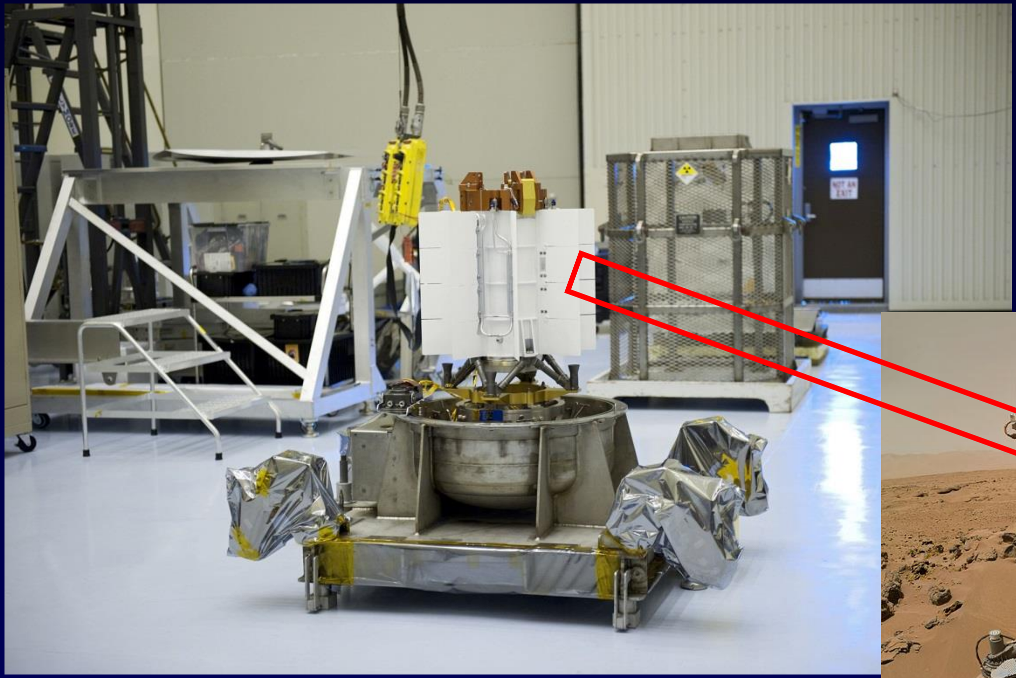
- First laser spectrum of chemical elements from ChemCam on *Curiosity*

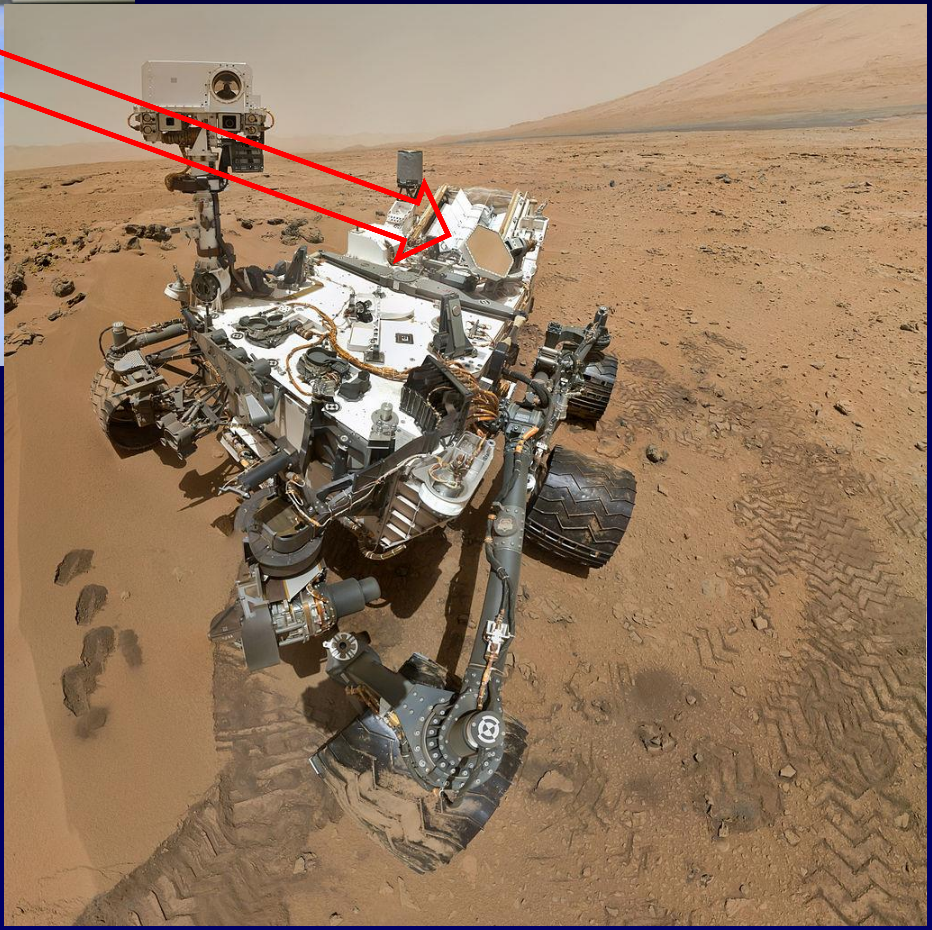


Multi-Mission Radioisotope Thermoelectric Generator (MMRTG)

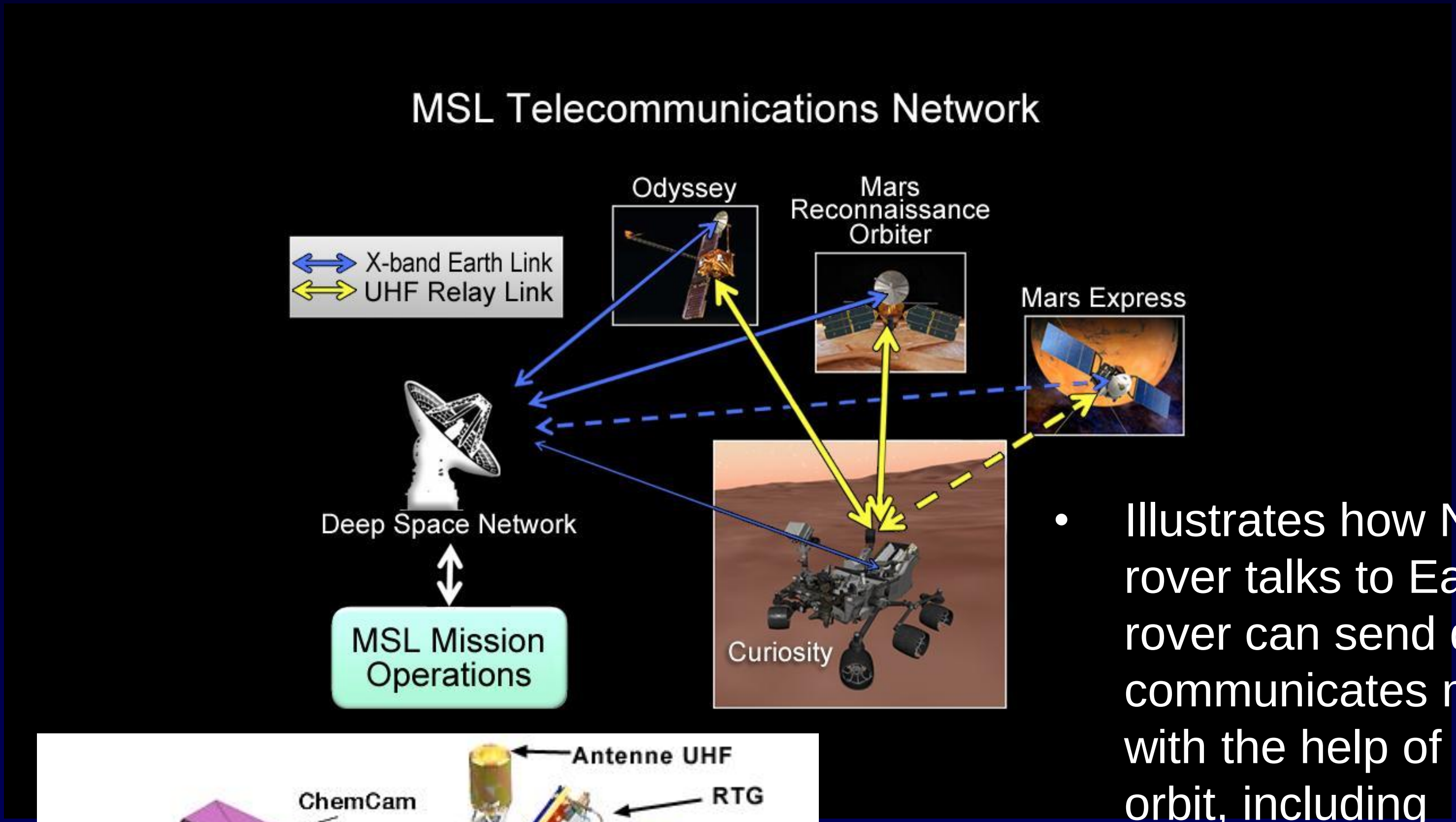


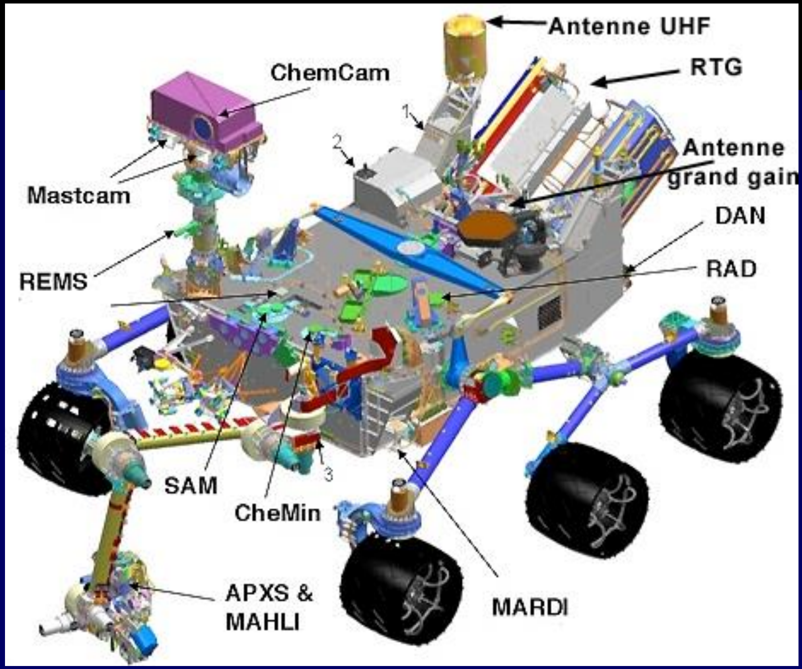
The Multi-Mission Radioisotope Thermoelectric Generator of Mars Science Laboratory





- The MMRTG design incorporates PbTe/TAGS thermoelectric couples (from Teledyne Energy Systems). The MMRTG is designed to produce 125 W electrical power at the start of mission, falling to about 100 W after 14 years.

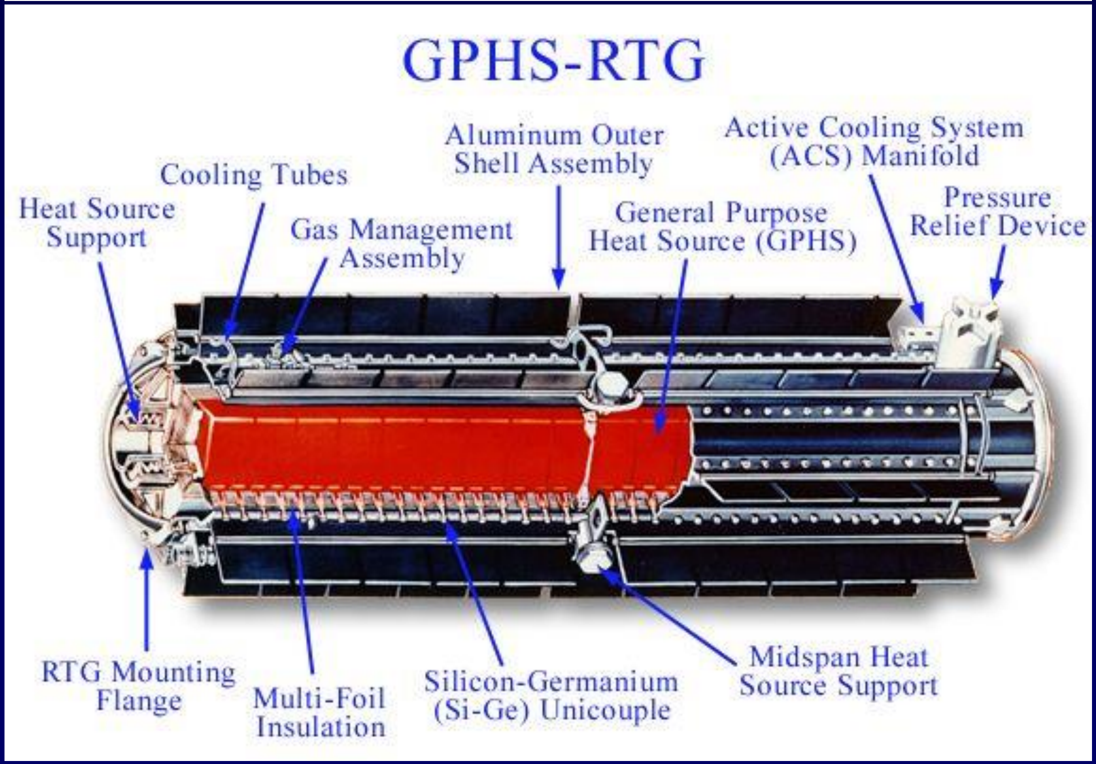




- Illustrates how NASA's Curiosity rover talks to Earth. While the rover can send direct messages, it communicates more efficiently with the help of spacecraft in Mars orbit, including
  NASA's Odyssey and Mars Reconnaissance Orbiter, and the European Space Agency's Mars Express.

Radioisotope Thermoelectric Generator - Satellite Type

Radioisotope Thermoelectric Generator



RTG Safety - Satellite Type Device

- Provide power through the natural radioactive decay of plutonium (mostly Pu-238, a non-weaponsgrade isotope). The heat generated by this natural process is changed into electricity by solid-state thermoelectric converters.

- RTGs are not nuclear reactors and have no moving parts.

- First, the fuel is in the heat resistant, ceramic form of plutonium dioxide, which reduces its chance of vaporizing in fire or re-entry environments.

- This ceramic-form fuel is also highly insoluble, has a low chemical reactivity, and primarily fractures into large, non-respirable particles and chunks.

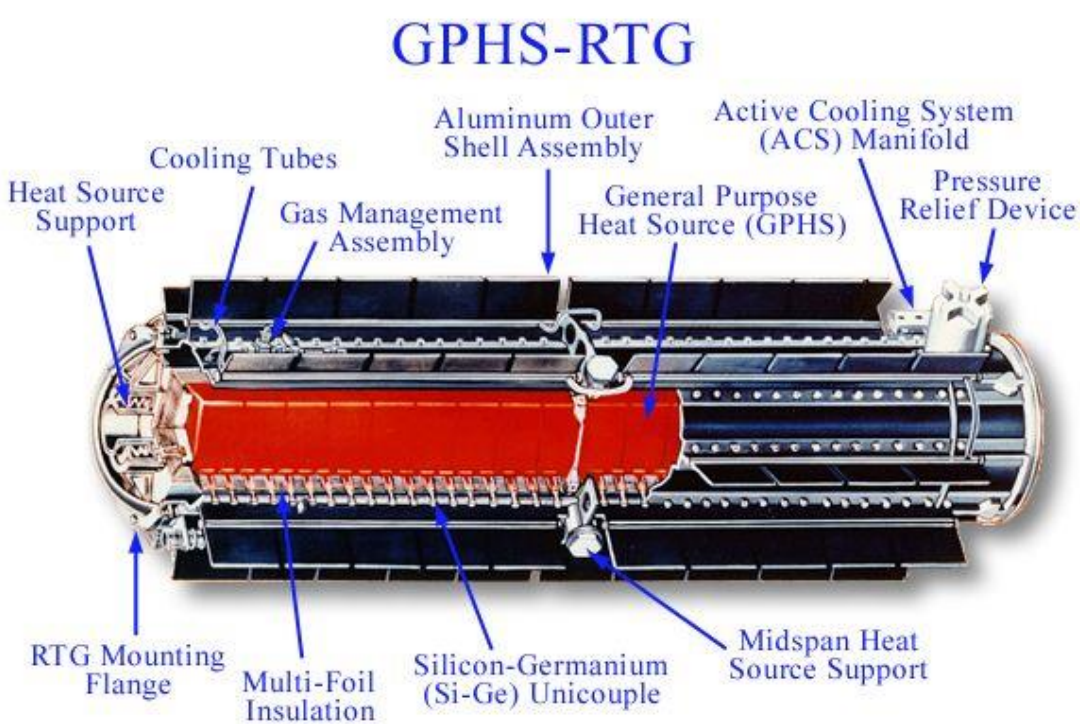
- These characteristics help to mitigate the potential health effects from accidents involving the release of this fuel.

RTG Safety

- Second, the fuel is divided among 18 small, independent modular units, each with its own heat shield and impact shell. This design reduces the chances of fuel release in an accident because all modules would not be equally impacted in an accident.
- Third, multiple layers of protective materials, including iridium capsules and high-strength graphite blocks, are used to protect the fuel and prevent its accidental release. Iridium is a metal that has a very high melting point and is strong, corrosion resistant and chemically compatible with plutonium dioxide. These characteristics make iridium useful for protecting and containing each fuel pellet. Graphite is used because it is lightweight and highly heat resistant.

Radioisotope Thermoelectric Generator - Satellite Type

Radioisotope Thermoelectric Generator



Radioisotope Thermal Generator - RTG

The advantages of RTGs over other systems include the following:

1. They provide independence of power production from spacecraft orientation and shadowing.
2. They provide independence of distance from the Sun (deep-space missions are possible).
3. They can provide low power levels for long periods of time.
4. They are not susceptible to radiation damage in the Van Allen belts.
5. They are suitable for missions with long eclipse periods, for example, lunar landers.

Radioisotope Thermal Generator - RTG

The disadvantages of RTG systems need also be considered, and include

1. They adversely affect the radiation environment of the satellite whilst in orbit. This will influence the spacecraft configuration significantly as may be seen from Figure 10.13, which shows the Galileo spacecraft. In this instance, the RTG needs to be deployed on a lengthy boom away from the main satellite bus.
2. Careful handling procedures are required during satellite integration owing to the radiation hazard posed by the radioactive source.
3. High temperature operation is required for efficient energy conversion. This impacts upon the thermal environment of the vehicle, and again on vehicle configuration.
4. RTGs are a source of interference for plasma diagnostic equipment that may be carried as part of the scientific objectives of the mission.
5. At the political level there has been increasing concern expressed at the inclusion of radioactive material on board a satellite. This is principally of concern because of the potential for such a source to be dispersed in the atmosphere, should there be a launch failure.

Radioisotope Thermal Generator - RTG

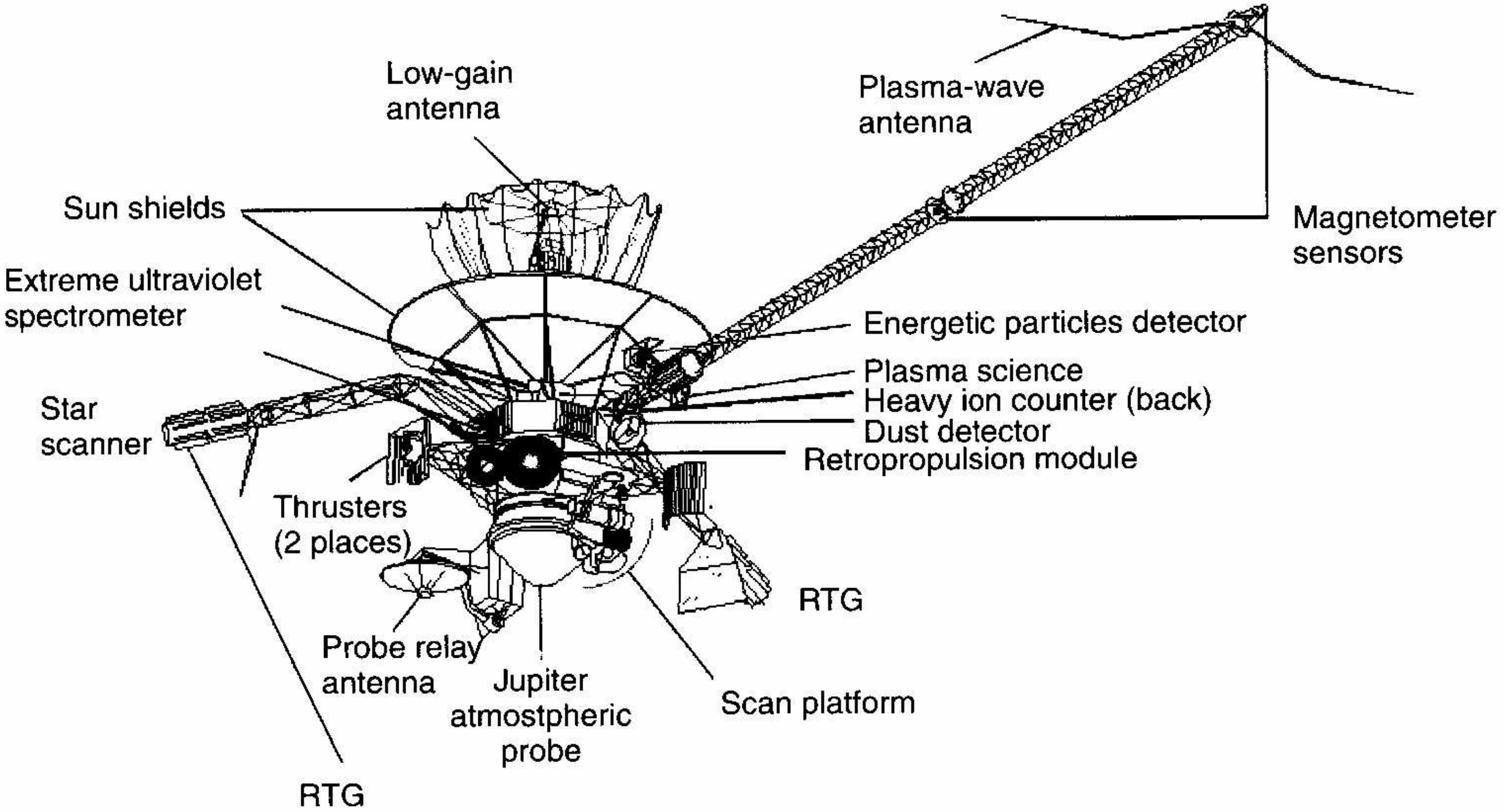


Figure 10.13 The Galileo spacecraft configuration, showing the position of the RTG sources (Courtesy of NASA/JPL/Caltech)

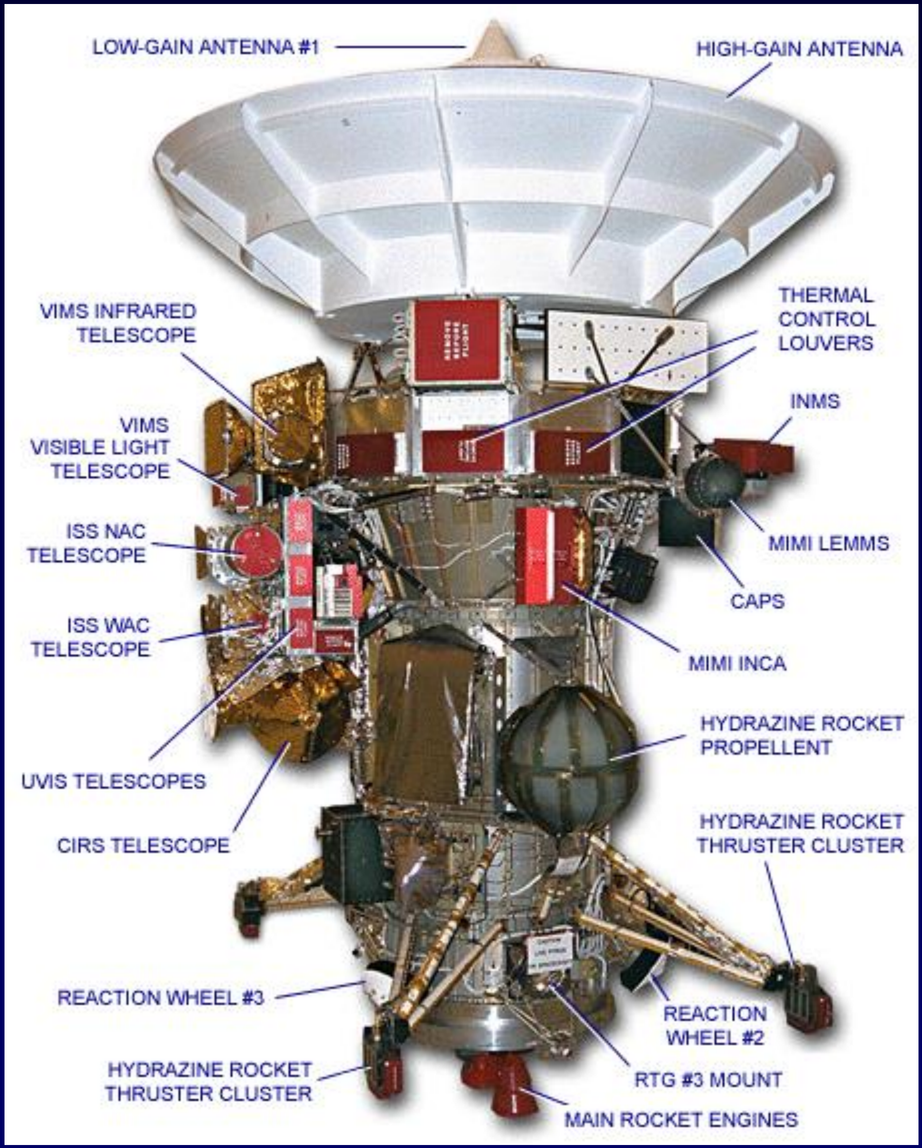
Radioisotope Thermal Generator - RTG

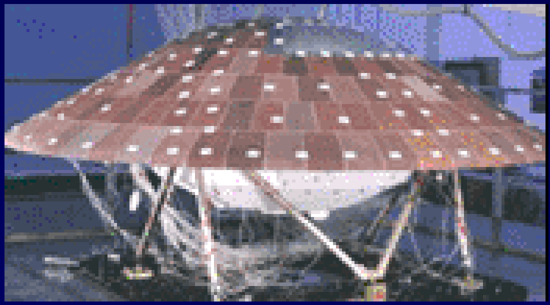
Table 10.5 RTG system performance

Name	Power (W)	kg/kW
Cassini (1997 launch)	628	195
Galileo probe/Ulysses (GPHS RTG, late 1980s)	285	195
Nimbus/Viking/Pioneer (SNAP 19, mid 1970s)	35	457
Apollo lunar surface experiment:		
SNAP-27, early 1970s	25	490
SNAP 9A, 1960s	73	261

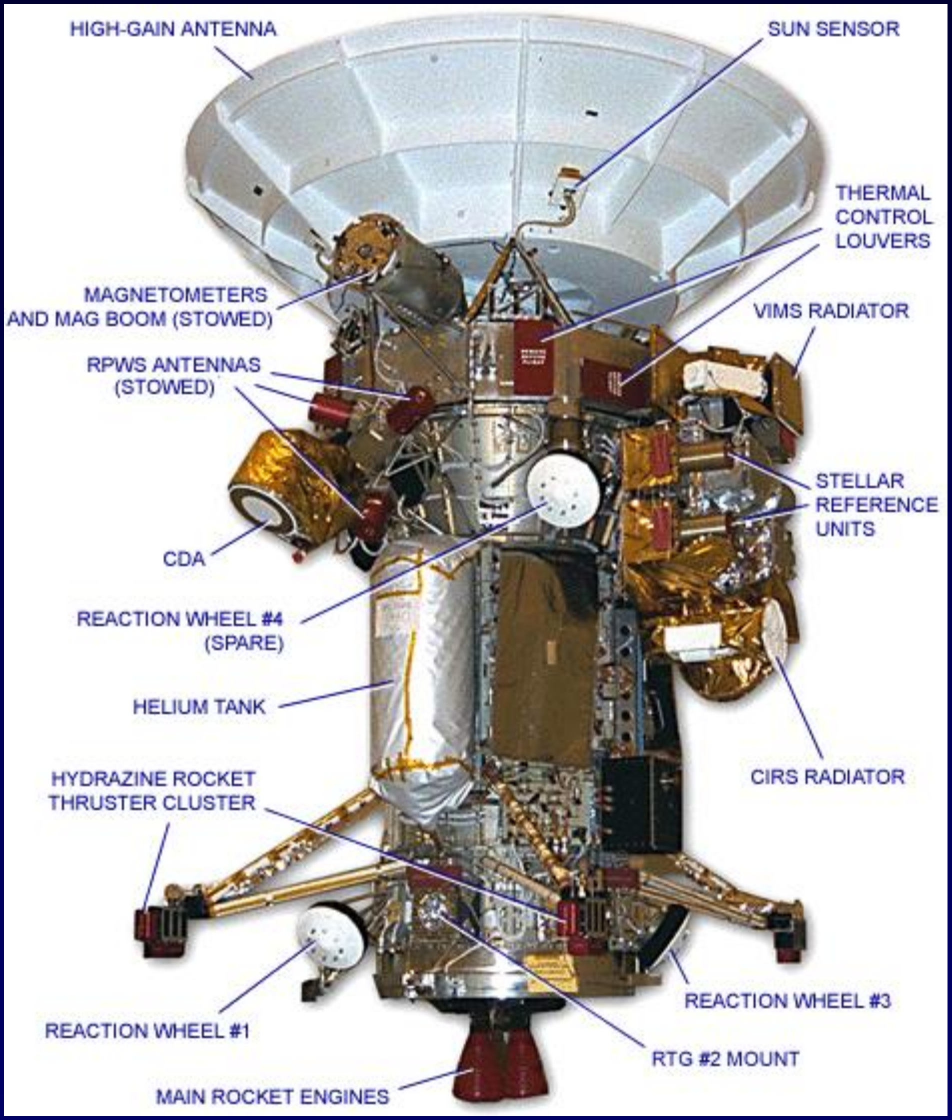
Cassini-Huygens Spacecraft

LOOKING AT THE MINUS-Y SIDE OF THE CASSINI SPACECRAFT

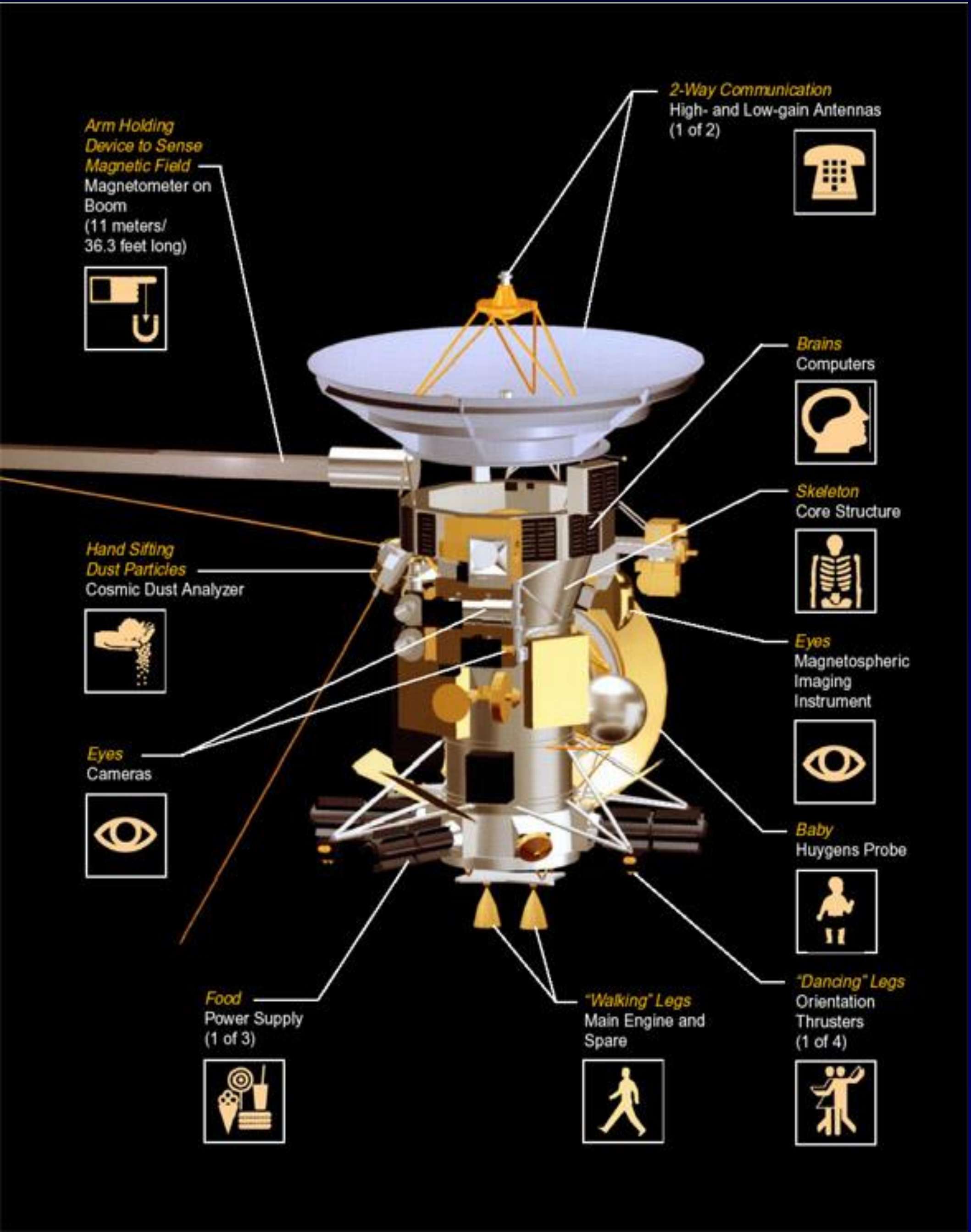




LOOKING AT THE PLUS-Y SIDE OF THE CASSINI SPACECRAFT



NASA Cassini to Saturn & ESA Huygens probe to Titan



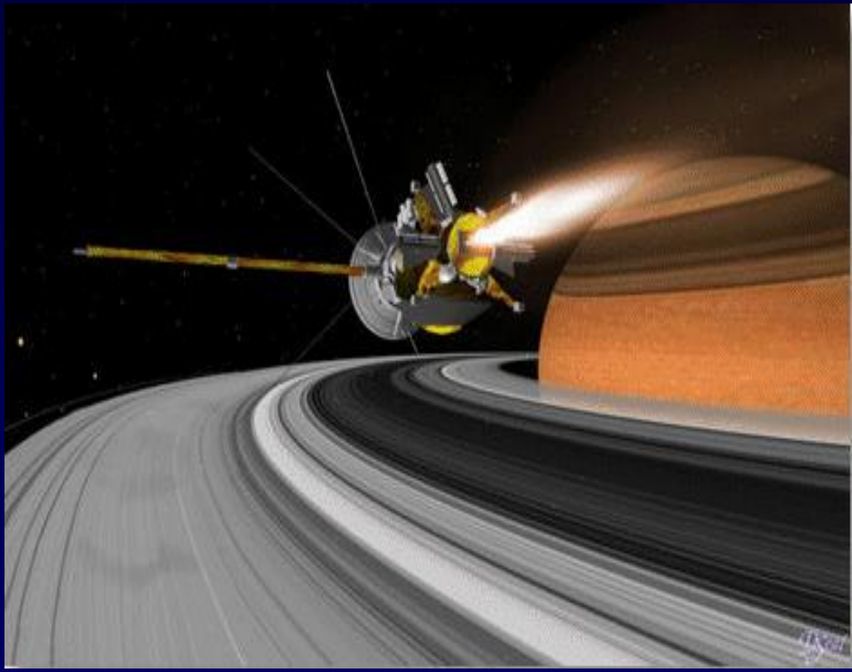


Image from Cassini

Fuel Cells

- Fuel cells provide the primary power source for the shuttle orbiter
- Originally they were designed as part of the Mercury, Gemini and Apollo US manned missions.
- Table 10.3 shows how their performance has evolved since the earliest days of manned space flight.
- A fuel cell converts the chemical energy of an oxidation reaction directly into electrical energy, with minimal thermal changes.
- The hydrogen/oxygen fuel cell has been used for space applications, a product of the reaction being water. This is clearly useful for manned missions. See figure 10.10

Fuel Cells

Table 10.3 Performance summary of fuel cells for space use

System	Specific power (W/kg)	Operation
Gemini	33	
Apollo	25	
Shuttle	275	2500 h at P_{ave}
SPE technology	110–146	>40 000 h
Alkaline technology	367	>3000 h
Alkaline technology	110	>40 000 h
Goal (lightweight cell)	550	

Note: SPE solid polymer electrolyte.

Hydrogen-Oxygen Fuel Cell

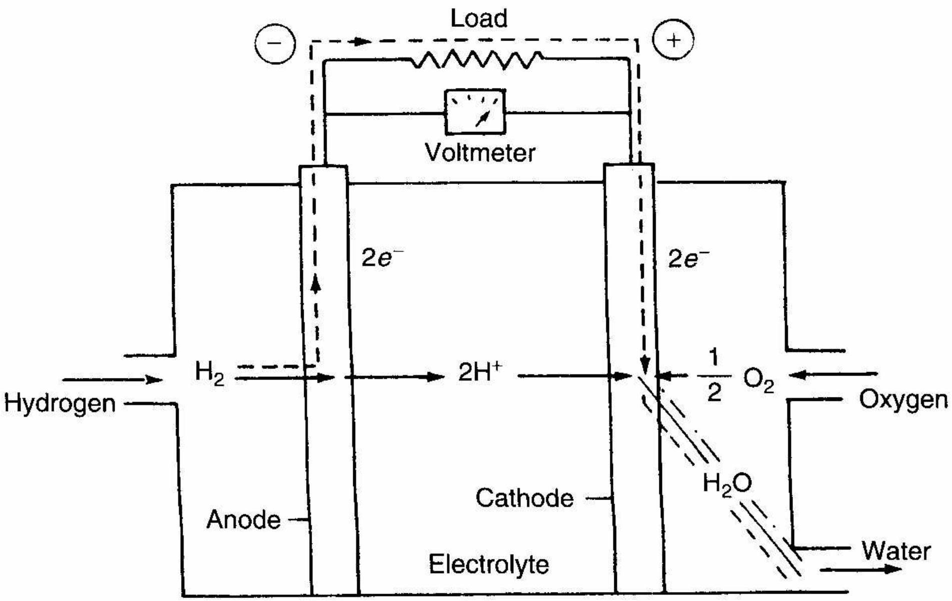


Figure 10.10 Schematic of a hydrogen/oxygen fuel cell. At the anode–electrolyte interface, hydrogen dissociates into hydrogen ions and electrons. The hydrogen ions migrate through the electrolyte to the cathode interface where they combine with the electrons that have traversed the load [2] (From Angrist, S. W. (1982) *Direct Energy Conversion*, 4th edn, Copyright Allyn and Bacon, New York)

Fuel cells – crudely = inverse electrolysis
Chemical fuels & oxidants stored external to cell.
Chemical reaction between hydrogen/solution and oxygen/solution:
At anode hydrogen dissociates → H+ and e
Hydrogen Ions migrate through solution to the cathode where they combine with the electrons that flow round in external circuit traversing the load.

The half cell reactions that describe the above cell are as follows:

Anode: $H_2(g) \rightarrow 2H^+ + 2e^-$ $\qquad V^0 = 0$

Cathode: $\frac{1}{2} O_2(g) + 2H^+ \;\; 2e^- \rightarrow H_2O(l)$ $\qquad V^0 = 1.23$ volts

Which for the cell yields: $H_2(g) + \frac{1}{2} O_2(g) \rightarrow H_2O(l)$

Fuel Cells

- The voltage that appears at the terminals of an ideal cell is given by

$$E_r = \frac{-\Delta G}{nF}$$

- Where ΔG is the change in Gibbs free energy occurring in the reaction, n is number of electrons transferred and F is the Faraday constant which is the product of Avagadro number and elementary charge; equals 9.65×10^4 C/mol
- For the reaction of the H_2/O_2 cell, two moles of electrons are transferred per mole of water formed and ΔG has the value of –237.2 kJ/mol at 25 deg C. The reaction takes place spontaneously
- Thus the reversible voltage of the ideal cell is:
- $237.2 \times 10^3 / (2 \times 9.65 \times 10^4) = 1.229$ V

Fuel Cells

- In practice this is not realised because there are various irreversibilities, termed polarisation losses
- Figure 10.11 shows a typical current-voltage curve for a hydrogen/oxygen fuel cell
- The magnitude of voltage drop is given by the Tafel equation:
- $\Delta V_{ACT} = a + b \ln J$
- Where J is the current density at the electrodes and a and b are the temperature dependent constants for the reaction/surface description
- Shuttle fuel cell uses matrix aqueous alkaline technology with a specific power of ~12kW, 275 W/kg; the start up time is 15 minutes as is the shutdown time.

Fuel Cell Performance

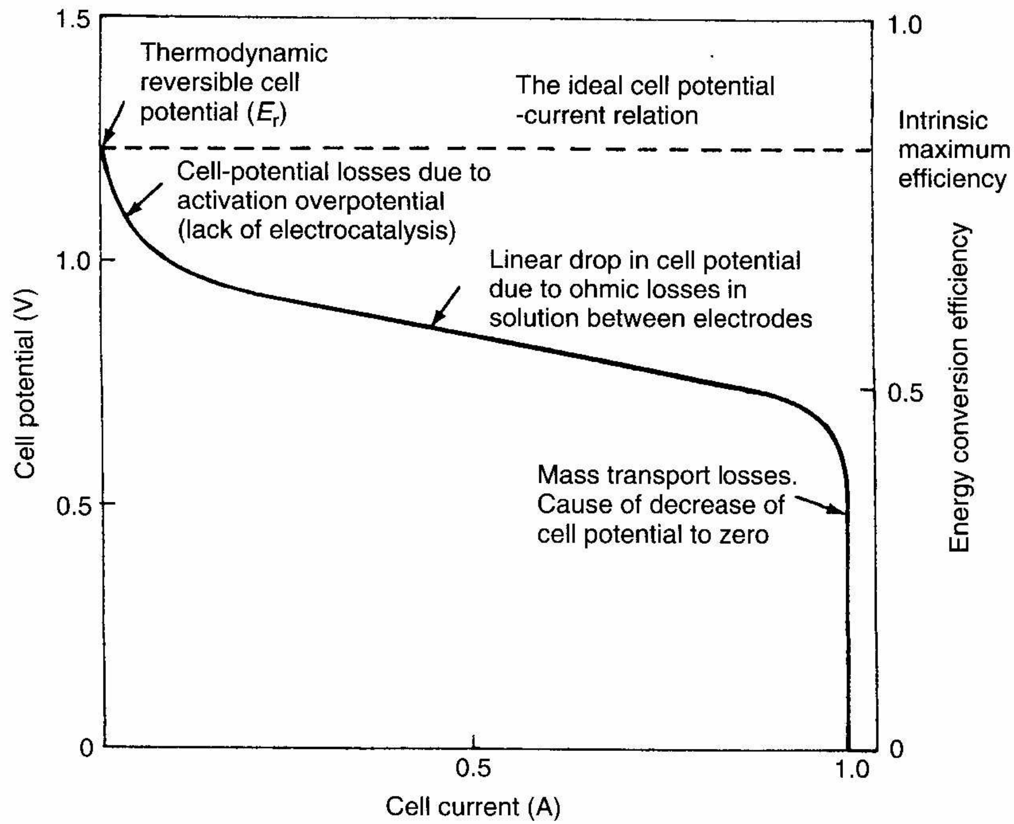
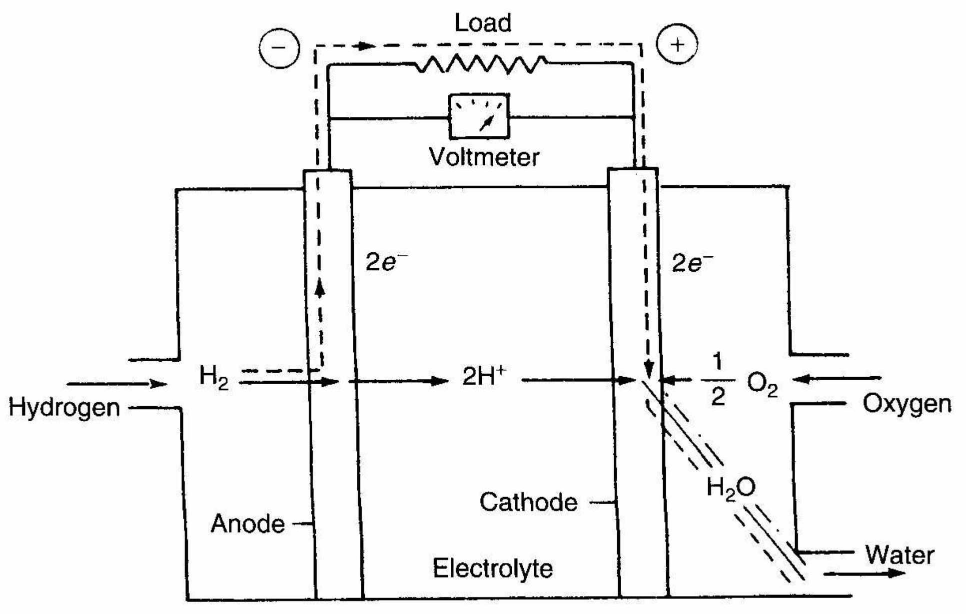


Figure 10.11 Typical cell potential and efficiency–current relation of an electrochemical electricity producer showing regions of major influence of various types of overpotential losses (Source [10])

Hydrogen-Oxygen Fuel Cell



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Which for the cell yields: $H_2(g) + \frac{1}{2} O_2(g) \rightarrow H_2O(l)$

Thermodynamic Properties

Table (4) Thermodynamic properties for the reaction that produces water from hydrogen and oxygen

T (K)	ΔH^o (kcal/mol)	ΔG^o (kcal/mol)
298	-57.80	-54.64
400	-58.04	-53.52
500	-58.27	-52.36

Faraday's constant F = 23.060 kcal / (volt-mole-elec)

For the hydrogen-oxygen cell operating in the standard state of 298 K and 1 atmosphere, the actual cell voltage is 0.75 V. What is the actual efficiency

What is the Voltage efficiency

$V^0 = 0$

Cell Efficiency

T (K)	ή = ΔG° / ΔH°	Carnot efficiency = $(T_H - T_C) / T_H$
298	0.94	0
400	0.92	0.26
500	0.90	0.40

The Actual Efficiency is $ή_{ac} = -nFV_{actual} / \Delta H^o$

$$= -2\ \frac{mole-elect}{mole}\ x\ 23.060\frac{kcal}{volt-mole-elect}\ x\ 0.75 volt / -57.8\frac{kcal}{mol}$$

$ή_{actual} = 0.598$

Voltage efficiency = V_{actual} / V^0 = 0.75 / 1.23 = 0.61

Some benefits

- Potable water produced
- Useful for short duration missions
- Very useful for shuttle missions- tiled surface prohibits solar cell array deployment.
- Fuel cells can provide astronauts with drinking water → better than the alternative – filtered urine!
- Very high efficiency

In the Shuttle Orbiter, a complement of three 12kW fuel cells produces all onboard electrical power; there are no backup batteries, and a single fuel cell is sufficient to insure safe vehicle return.

Each fuel cell is a self-contained unit 14 x 15 x 45 inches, weighing 260 pounds. They are installed under the payload bay, just aft of the crew compartment. The power plants are fuelled by hydrogen and oxygen from cryogenic tanks located nearby. Each fuel cell is capable of providing 12 kW continuously, and up to 16 kW for short periods.

Each power plant contains 96 individual cells of the alkaline (KOH) electrolyte technology which are connected to achieve a 28 volt output. The cells are over 70% efficient (a typical combustion engine is only about 25% efficient); this high efficiency and light weight led NASA to select fuel cells to power the Space Shuttle Orbiter.

Other Primary Power Systems

- Nuclear Fission systems are similar to conventional ground based nuclear power stations, in that uranium-235 is used as a heat source.
- In space systems, this is used to drive a thermoelectric converter
- Solar heat systems can be used to drive a heat engine and then a rotary converter to electricity, or used as a heat source for a thermo electric converter
- Solar dynamic systems have had the greatest concentration of effort for the ISS. Design studies have shown that their end to end conversion efficiency is 25% greater than for current photovoltaics. But smaller than triple junction photovoltaics
- This results in a reduced need for deployed collection area by about 25%, and consequently in reduced aerodynamic drag for LEO satellites

SP-100
Space Reactor
Power System

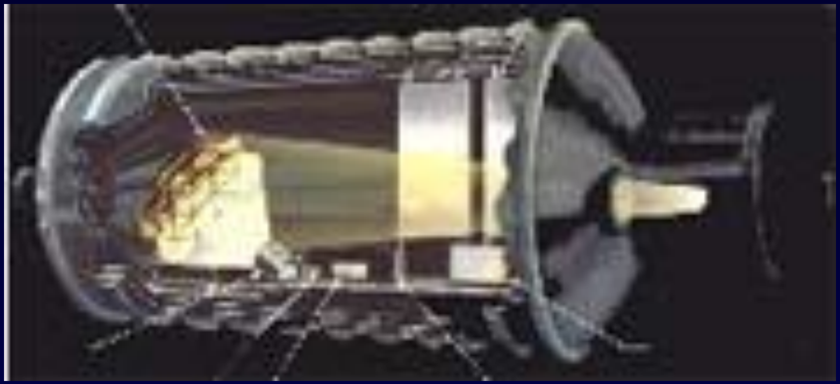
Multimegawatt Space
Nuclear Power System

DIPS
(Dynamic Isotope
Power System

In nuclear, Boeing's heritage includes development of space nuclear auxiliary power systems (SNAP 8 and SNAP 10A programs), high space power reactor systems (100 kWe SP-100 and 5 MWe Multi-Megawatt programs), and dynamic isotope power systems (2.5-to-6 kWe DIPS programs).

Solar Dynamic Systems

- In the original concept for the ISS, primary power for the initial in-orbit capability was to be 75 kW, derived from photovoltaics. Power expansion was then assumed to be provided by solar dynamics in two units of 25 kW
- Solar dynamic systems, resulting in less drag, lead to lower fuel usage for orbit maintenance. This reduces the cost of station operation, principally by reducing the demand for Shuttle refuelling flights. Cost savings of several billion dollars have been identified
- Brayton cycle shown in figure 10.14; working fluid for this all-gas phase cycle is helium and xenon in such a proportion that the equivalent molecular weight is 40.
- Storage of energy within the concept uses the latent heat of fusion for a lithium fluoride/calcium fluoride mixture. This phase change occurs at 1042 K. Storing energy in this way provides mass savings compared to battery technology.



Solar Dynamic
Power Module

Solar Dynamic Ground Test
Demonstrator (SDGT)

Stirling Engine for
Deep-Space
Missions

- High density, dynamic power conversion systems (e.g., Brayton and Stirling power conversion cycles),
- High voltage electric power management and distribution systems (e.g., 115-173 Vdc primary and regulated 113-126 Vdc secondary power systems for the ISS, with >94% power conversion efficiency, solid state power switching and fault protection capability)

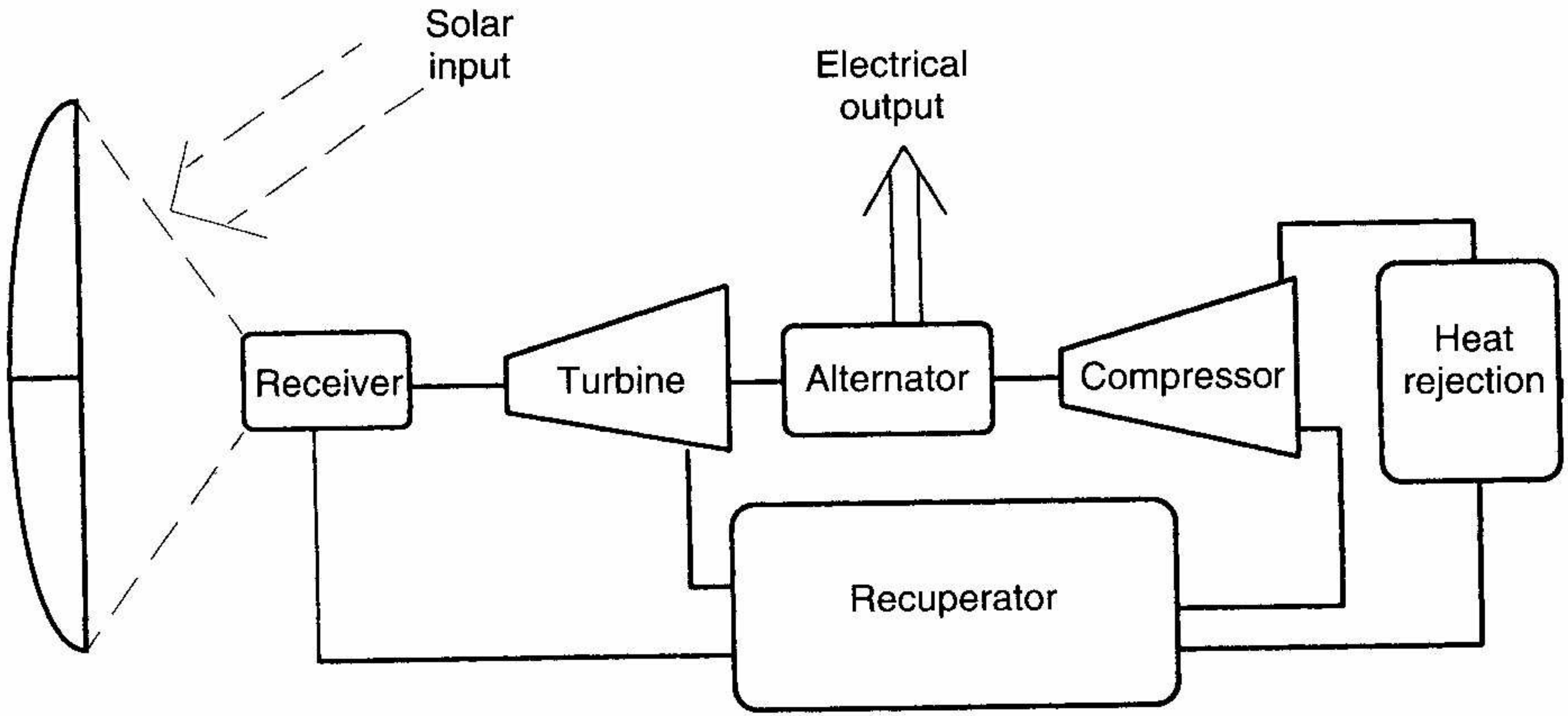


Figure 10.14 Solar dynamic Brayton cycle

Secondary Power Systems - Batteries

- Batteries have been used extensively for the secondary power system, providing power during periods when the primary one is not available
- As a backup for a solar array this means that the batteries must provide power during eclipses, and that the array must recharge the batteries in sunlight.
- In GEO operations eclipse lasts for a maximum of 1.2 hrs. with more than 22 hrs of sunlight available in each orbit, a trickle charge solution is possible
- In LEO the spacecraft may be in eclipse and thus require battery power for 40% of each orbit. Typically 5000 to 6000 charge/discharge cycles of the battery will be required per year
- This results in the array-power sizing needing to be nearly twice the nominal load requirement

Secondary Power Systems - Batteries

- In summary LEO operations require a large number of low-depth discharges, whereas in GEO a few deep discharges suffice
- This influences battery type, resulting in the present trend of using:
- Nickel-cadmium (Ni-Cd) or Silver-Zinc (Ag-Zn) cells for LEO operation
- Nickel-hydrogen cells (Ni-H_2) cells for GEO operations; Li Ion
- Cell cycle life, specific weight (kW h/ kg) and volume (kW h/m^3) all influence the acceptability of a particular battery technology
- Work on more exotic materials, for example, Li-SO_2 is continuing and alternative technologies continue to be implemented on spacecraft.

Secondary Power Systems - Batteries

- A Li-SO_2 battery provided power to the Huygens probe, after separation from the Cassini Saturn orbiter
- For this rather specialised mission, the probe is in hibernation for 7 years
- The battery is then required to provide power during a low power coast for 22h, and then a high power load for 2.5h during descent
- For the final 30 minutes of this period the probe operates from the surface of Titan at –180 deg C
- For this, the overall battery unit contains five individual battery units. Each battery consists of two modules of 13 Li-SO_2 cells in series. Each cell has a capacity of 7.5 A-h

Secondary Power Sources - Batteries

Table 10.6 Performance of battery technologies for space use [14]

Type	Specific energy (W h/kg)
Ni–Cd	39
Ni–H_2	52
Ag–Zn	60
Ni–MH	60
Li–Ion	80
Li–TiS_2	125
Na–S	150

Battery Conditioning

- Parameters of critical importance are the charge/discharge rate, the depth of discharge (DOD), the extent of overcharging and the thermal sensitivity to each of these parameters.
- Figure 10.15 summarises some of the available data on Ni-Cd batteries.
- Figure 10.16 demonstrates the loss of charge capacity over several cycles.
- Battery reconditioning before an eclipse season is regularly used for GEO orbit spacecraft
- It is noted that if a battery is completely discharged, then capacity may be regained

Table 10.7 Hubble space telescope and Intelsat VII $Ni-H_2$ battery summary

Parameter	HST	Intelsat VII
Specific energy (W h/kg)	57.14	61.26
Capacity (A h)	96	91.5
Cell dimensions:		
Diameter (cm)	9.03	8.89
Length (cm)	23.62	23.67
Terminal/terminal (cm)	24.66	29.67
Cell mass (kg)	2.1	1.867

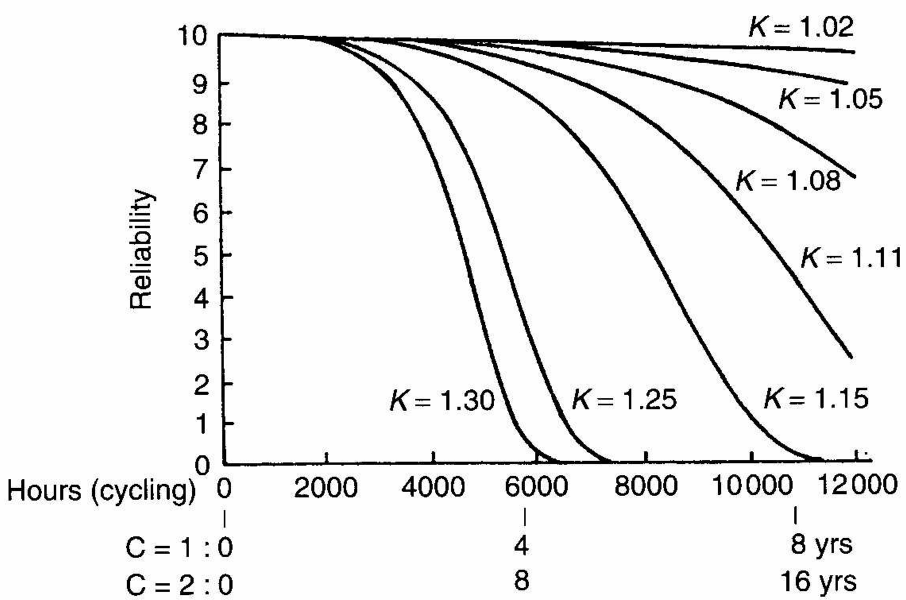


Figure 10.15 Ni–Cd cell reliability as a function of overcharge factor. Hours cycling is related to operation for two cases: $c = 1$, charge rate is battery capacity $\frac{1}{20}$ A/A h; $c = 2$, charge rate is battery capacity $\frac{1}{10}$ A/A h [17] (Reproduced by permission of European Space Agency and P. Montalenti)

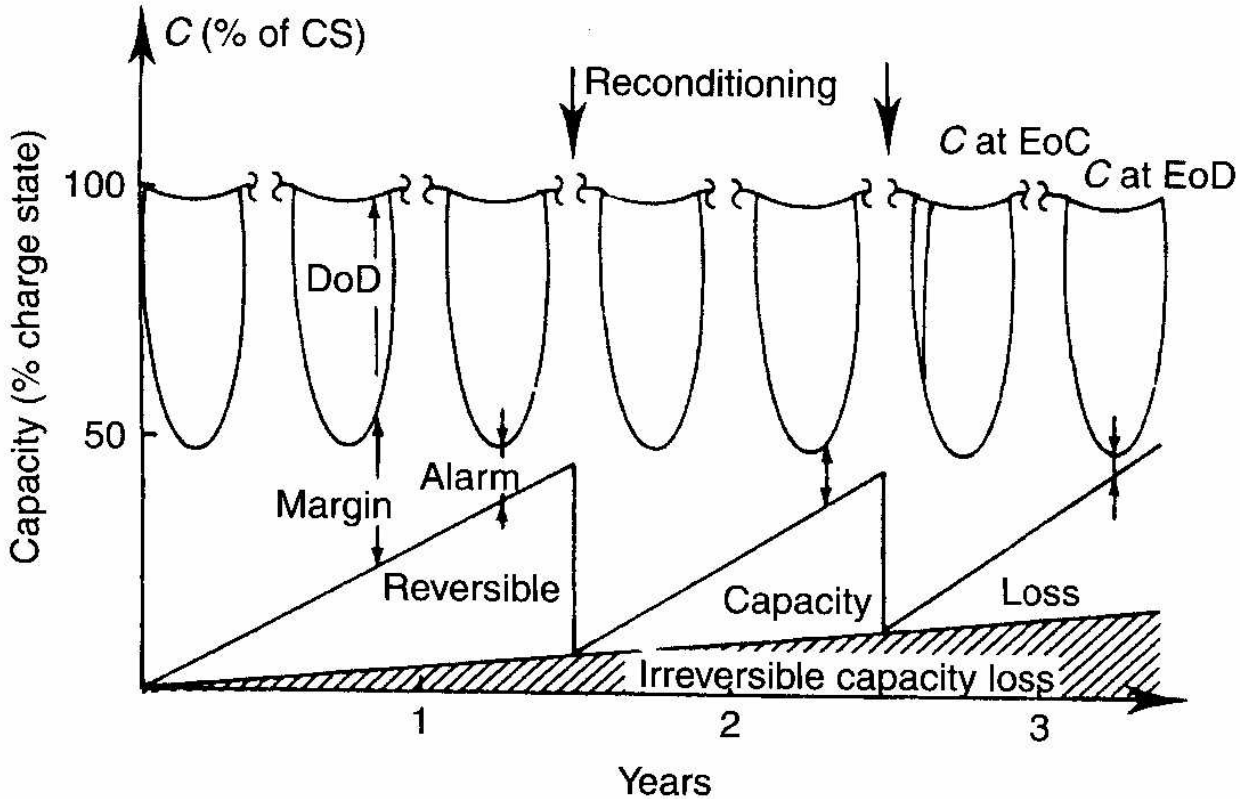


Figure 10.16 Battery reconditioning via complete discharge to improve battery capacity. Both reversible and irreversible capacity loss occurs [17] (Reproduced by permission of European Space Agency and P. Montalenti)

Power Conditioning

- **Orbital Day**

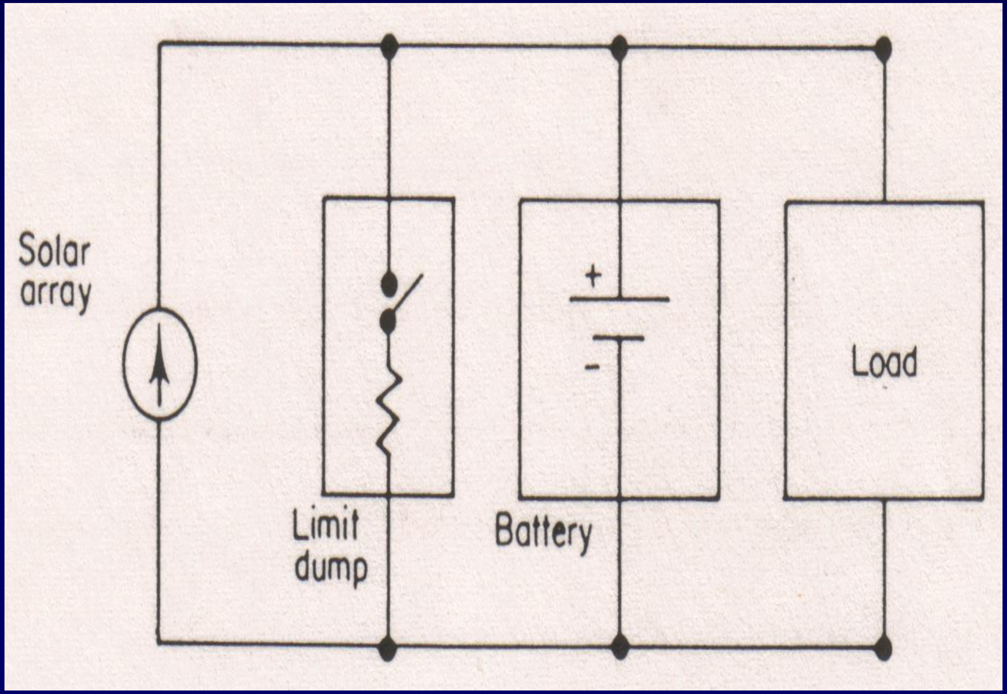
  Solar Cells provide power to charge batteries & power all systems

- **Orbital Night**

  Storage battery supplies all system power. Night can be up to 70min/day GEO orbit, or 45% of 90minute polar LEO orbit

Solar Cells designed to provide excess voltage because of variations with temperature, degradation with time (radiation & micrometeorites)

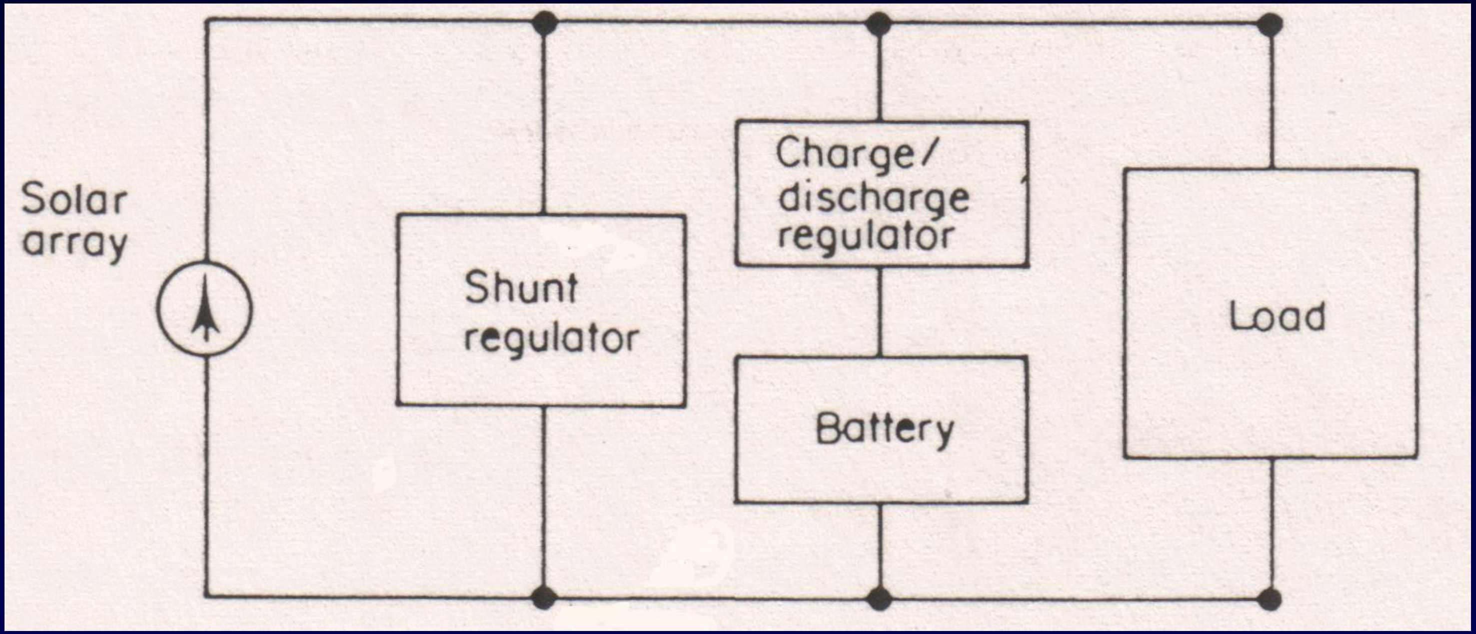
→ Need 1) Voltage regulation + 2) Load (resistor) to dump excess current



1. Simple Unregulated BUS

Battery in parallel with solar array, defines operating voltage of array, and of user load.
Current dump needed for excess.
Simple low mass system but system equipment subject to voltage variations with charge/discharge of battery & solar array bias not necessary on efficient 'knee'

Regulated BUS



Regulated BUS- Shunt regulator defines solar array voltage- optimises on ‘knee’
Batteries charged/discharged under control of regulator
Steady load voltage.
More mass, less power available, less reliable

General Power Aspects:
Spacecraft system loads include individual current protection / limit circuits and limits on switch on ‘in-rush current’, etc
Main bus often 28V so need many voltage conversion circuits (e.g. logic 5,3.3,1.5V)
DC-DC convertors: 28V DC→oscillators (kHz/MHz)→AC→transformer→rectify→DC

Power Management, Distribution and Control

- An ac bus is sometimes used to augment the dc one. The hybrid system can provide mass savings due to both the simplicity of conversion from ac to dc to a variety of dc levels, and also the fact that it is possible to run the power distribution harness at higher voltages if an ac supply is used.
- An ac distribution is mainly applicable to a high power spacecraft and where a large number of dc voltages are required at the equipment level.
- The Shuttle Space lab uses an ac bus providing three phase sinusoidal voltage at 400 Hz

Power Management, Distribution and Control – Primary Units

- **Array regulator** –
  - As the power available from a mission, or power demanded by the payload varies, it is possible to switch in or out segments of the solar array
  - Switching out is achieved by grounding the individual segments in the shunt dump module
  - The array itself can be structured into various modules, sometimes called solar power assemblies (SPA)
  - The voltage sensing that is used to control the shunt dump module is termed the mode control unit (MCU)

Power Control and Distribution Unit (PCDU)

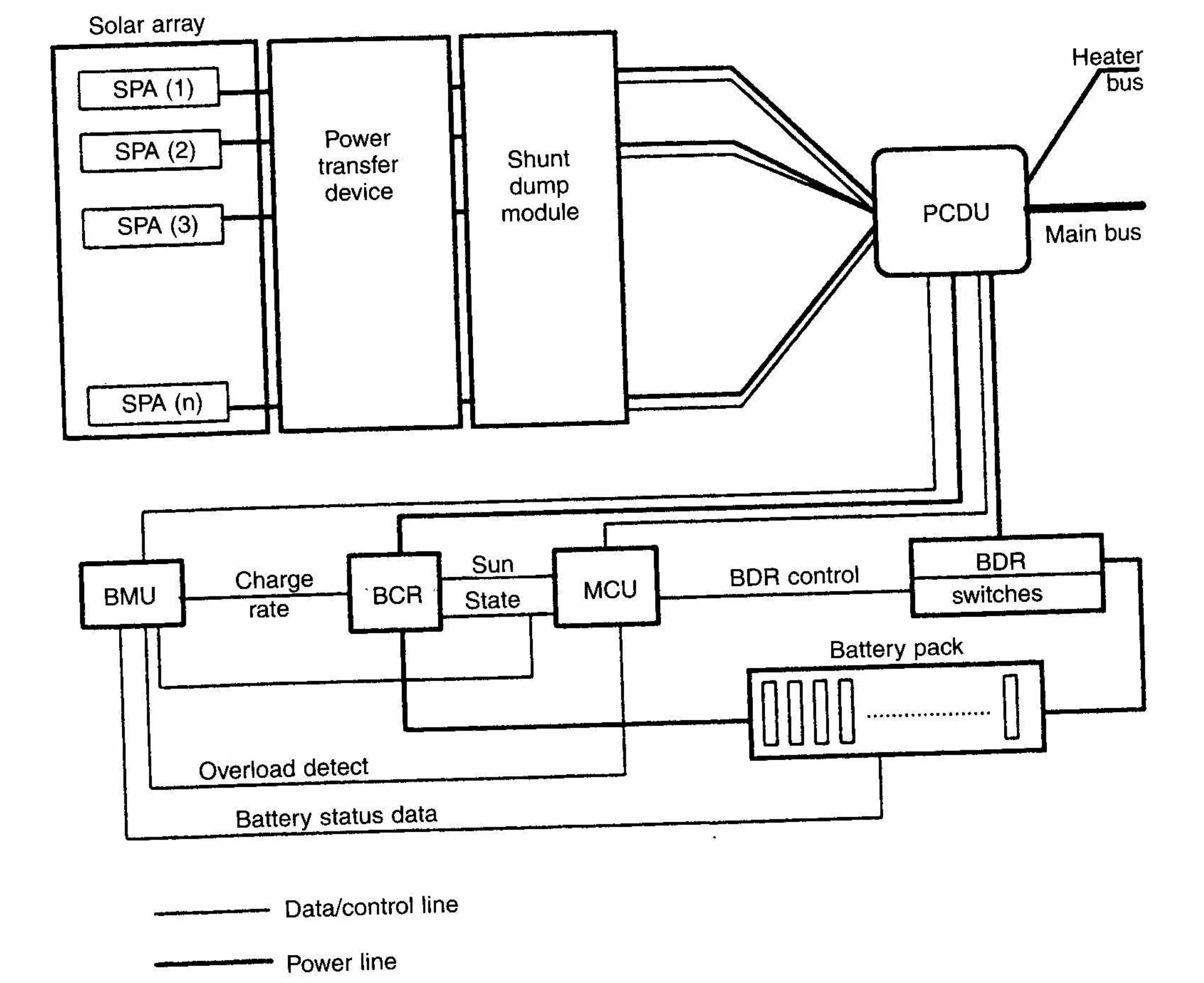


Figure 10.18 Power system layout

Power conversion unit (PCU)

The voltage sensing that is used to control the shunt dump module is termed the mode control unit (MCU)

Battery management unit (BMU)
Battery charge regulator (BCR)
Battery discharge regulator (BDR)

Power Management, Distribution and Control – Primary Units

- **Battery Control - Three units are associated with battery control**
  - Battery management unit (BMU)
  - Battery charge regulator (BCR)
  - Battery discharge regulator (BDR)
- **The BMU's functions are to monitor the battery's temperature and voltage as well as individual cell voltages, pressures and temperatures. It also provides control inputs to the charge regulation of the batteries, carried out by the BCR**
- **The principal function of the BCR is to provide a constant current charge of the battery during sunlight operation**
- **The BDR supplies a constant current to the spacecraft bus during eclipse operation.**
- **Control of this current is derived from the MCU, typically with further protection from the BMU**

Power Management, Distribution and Control – Primary Units

- **Power Control and Distribution Unit (PCDU) –**
  - This unit provides monitoring and protection for the bus current
  - Protection is normally achieved either by current limiting or by fusing, the latter usually requiring a redundant path to be switched into operation, normally from ground control.
- **Power conversion unit (PCU) –**
  - This unit supplies the individual voltage/current characteristics required for loads
  - The typical voltage outputs eg ±15V and 5V; will be regulated using solid-state switches that are pulse width modulated
  - This unit must also be able to cope with transient protection for over and under voltage and in-rush current limiting when units are switched on or off.

Power Conditioning Unit (PCU) for Satellites

5 kW

AIRBUS DEFENCE & SPACE

High Electric Power System Management and Distribution: With the experience gained on ISS, Boeing has developed a highly reliable Power Conditioning Unit (PCU), which is plug-n-play in design, scaleable and modular in construction, high power (3-12 kW), high voltage (28-120 Vdc), and passively cooled. The PCU provides shunt control for Solar Arrays, NiH2 or Li-Ion battery charge and voltage management, bus voltage regulation, solid state power switching and fault protection, and control interfaces to spacecraft computer. The PCU is also radiation and SEU tolerant and can be tailored for a variety of high power satellite applications.

Power Budget

- **The three critical issues that need to be considered are:**
  - The orbit parameters
  - The nature of the mission – communications, science or other
  - Mission duration
- **The orbit will define the duration of the eclipse periods, which together with the number of eclipses anticipated, will define the battery requirements, Clearly in deep space missions an investigation of the most appropriate primary power source will be influenced by the orbit**
- **The nature of the mission will have a significant impact on the types of load expected.**
- **Thus for a comms satellite, it can be anticipated that the primary load will be required at all times. The power demand in eclipse may well exceed that in sunlight, owing to the need both to operate the payload and to meet the additional burden from active, or power augmented elements, in the thermal control system (TCS).**
- **Navigation and broadcast satellites will have similar requirements to these. In contrast, a remote sensing spacecraft during eclipse may well not require the whole payload to be operational, particularly there are passive optical instruments.**

Power Budget

- **The mission duration will provide a major influence on the degradation of the power system. The two most significant influences are:**
  - The total radiation dose expected which may determine, for a satellite carrying a solar array, the amount of shielding required, and hence influence the specific mass of the power system
  - The number of eclipse cycles. Solar cell failure through open-circuit losses will clearly increase with the increased number of thermal cycles, driven by the entry to and from eclipse

System Efficiencies

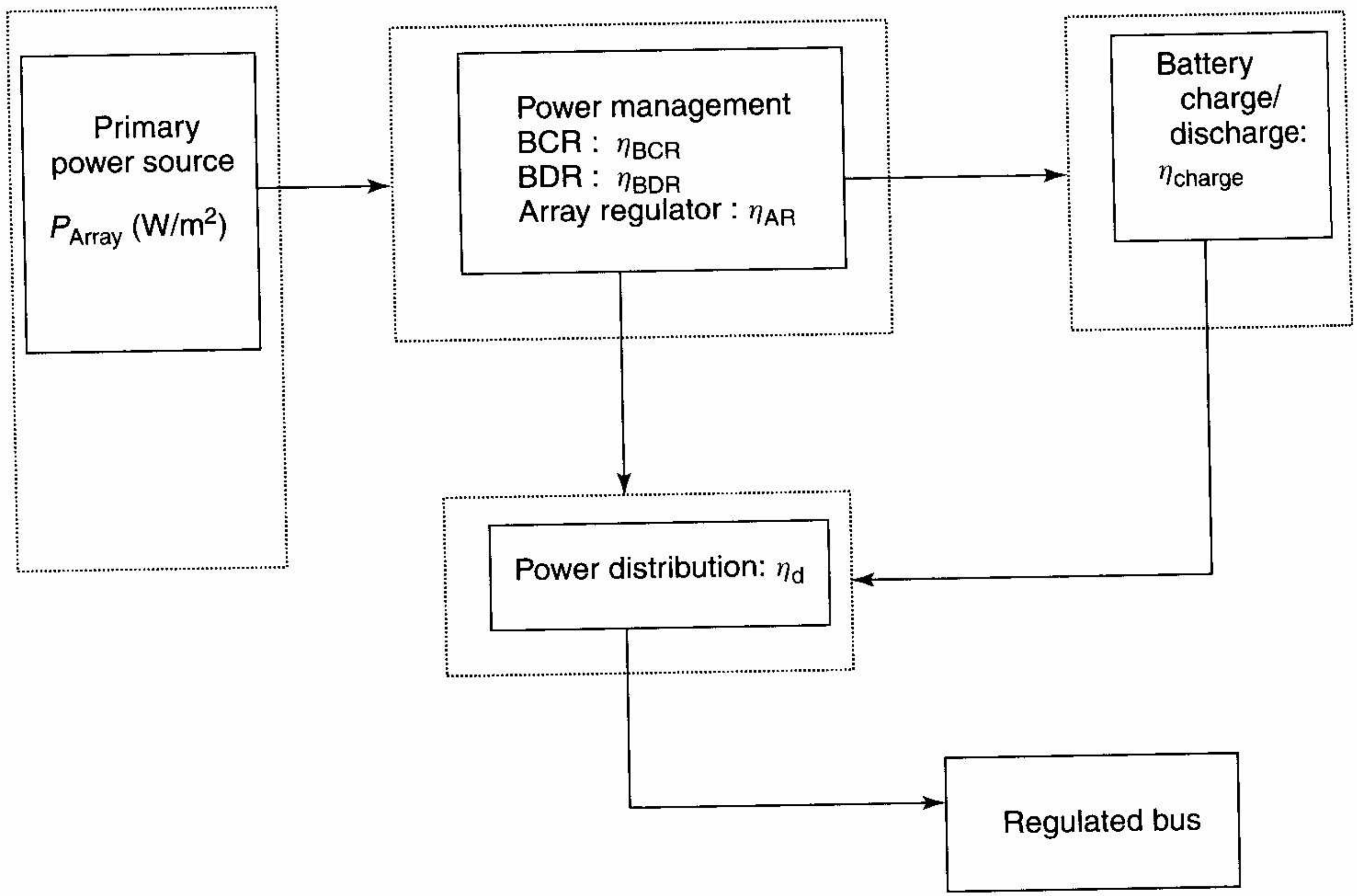


Figure 10.19 Definition of system efficiencies

Power Budget Evaluation

- For each sub-system, data must be provided for both the sunlit orbit phase – average value P_{sun} and the eclipse phase having an average value $P_{eclipse}$.
- Since the satellite must be provided with power throughout the mission, this specifies the EOL requirement.
- A simplified block diagram for the power system is shown in figure 10.19, in which the efficiencies of various components are also identified.
- Assuming that the orbit period is τ with the time spent in the sunlight τ_{sun} and the time spent in eclipse $\tau_{eclipse}$ and then the power required from the array to meet the eclipse load is clearly given by P_{charge} , where:
- $P_{charge}\,\tau_{sun} = 1/\eta\ P_{eclipse}\,\tau_{eclipse}$ ………………………5
- $\eta = \eta_{BDR}\,\eta_{BCR}\,\eta_{AR}$

Array Regulator (AR)
Battery charge regulator (BCR)
Battery discharge regulator (BDR)

Power Budget Evaluation

- The total power required to be available from the array is thus approximately given by:
- $P_{array} = P_{sun} + P_{charge}$ ………………………………6
- In the limit, if the efficiencies are equal to unity, from 5 and 6 that if the eclipse power demand is the same as in the sunlight then the array power is simply given by:
- $P_{array} = P_{sun}\, \tau / \tau_{sun}$ …………………………………..7
- Thus in LEO where the fraction of the orbit in eclipse is large, typically 30 min out of 90 min, τ / τ_{sun} =1.5 The array power needs to be 50% in excess of the bus load
- In GEO where a maximum eclipse duration of ~ 70 min in the 24 hour orbit arises. Over sizing of the array amounts to only 5%

Power Budget

Table 10.8 Typical structure of a power budget

Subsystem	Peak power	Sunlight power P_{sun}	Eclipse power $P_{eclipse}$	Intelsat VIIa (%)	Average GEO comms. satellite (%)
AOCS **(Altitude and Orbital Control)**				5.0	3.6
Power				10.4	11.2
Thermal control				4.9	6.4
Comms.				n/a	n/a
Data handling				0.6	1.6
Payload				79.1	77.2
Average total power				100	100

Power Budget

- If the orbit period is τ hours. The battery-stored energy will be given approximately by E_B (W-hrs)
- $E_B = P_{eclipse}(\tau - \tau_{sun})/(\eta_{charge}DOD)$……………………………8
- Where DOD is the depth of discharge of the battery
- The battery mass can be estimated by dividing the stored energy (W-hrs) by the energy density (W-hrs/kg) for the chosen battery technology
- We can modify (5) to obtain the total energy required from the array. This may be written in the form ε_{array} where:
- $\varepsilon_{array} = P_{array}\,\tau_{sun}$ $= \frac{1}{\eta_{sun}}\left(\sum_{i=1}^{k} P_i t_i\right) + \frac{1}{\eta_{ecl}}\left(\sum_{i=k+1}^{n} P_i t_i\right)$ ……………………….9
- In which the P_i , i = 1, …….n, gives the typical power profile for payload and subsystem operation throughout the orbit. A typical profile is illustrated in figure 10.20

Orbit Power Profile

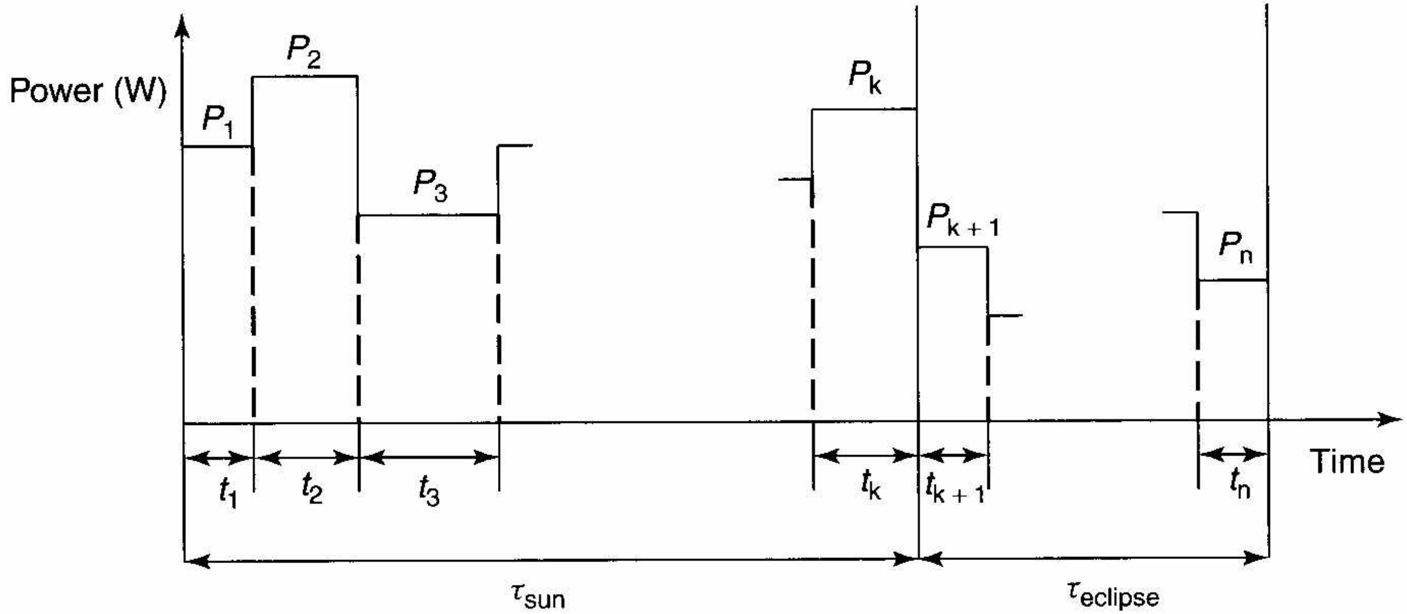


Figure 10.20 Approximate power profile for payload and subsystem operation throughout the orbit

Power Budget

- The battery charge requirement during sunlight is explicitly excluded from this profile, since the eclipse profile (second term in equation 9) is equivalent to the battery charge energy (see equation 5)
- For the purposes of a first estimate of the array size, the efficiency factors have typical values of η_{sun}~ 0.8, from array to loads, and η_{ecl}~ 0.6, from batteries to loads
- In terms of calculating the array size , allowances need to be made for any pointing angle off-set of the array relative to the sun line. Hence the array area is given by A_{array} where
- $A_{array} = P_{array}/(S \cos\delta\theta\ \eta_{cell}\ \eta_{packing}(1-D))$…………………..10
- S is the solar flux(~1400W/m^2 in near earth orbit); $\delta\theta$ is the array pointing error with respect to the sun, which is typically of the order of 1^o , but this depends on the mission; η_{cell} is the solar cell efficiency; $\eta_{packing}$ is the cell packing efficiency that is typically 0.9; D is the array degradation factor over the spacecraft lifetime
- Loss factors due to radiation damage, micrometeorite damage and battery hysteresis must be included in calculations. Loss factors may be as high as 25%

National Aeronautics and Space Administration
NASA
Juno Spacecraft
JunoCam
Ultraviolet Spectrograph (UVS)
Jovian Infrared Auroral Mapper (JIRAM)
Plasma Waves Instrument (WAVES)
Juno's Instruments
Gravity Science and Magnetometers
Study Jupiter's deep structure by mapping the planet's gravity field and magnetic field
Microwave Radiometer
Probe Jupiter's deep atmosphere and measure how much water (and hence oxygen) is there
JEDI, JADE and Waves
Sample electric fields, plasma waves and particles around Jupiter to determine how the magnetic field is connected to the atmosphere, and especially the auroras (northern and southern lights)
UVS and JIRAM
Using ultraviolet and infrared cameras, take images of the atmosphere and auroras, including chemical fingerprints of the gases present
JunoCam
Take spectacular close-up, color images
SPACECRAFT DIMENSIONS
Diameter: 66 feet (20 meters)
Height: 15 feet (4.5 meters)
Gravity Science
For more information:
missionjuno.swri.edu &
www.nasa.gov/juno
Jovian Auroral Distributions Experiment (JADE)
Microwave Radiometer (MWR)
Jupiter Energetic-particle Detector Instrument (JEDI)
Magnetometer
National Aeronautics and Space Administration
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California
www.nasa.gov

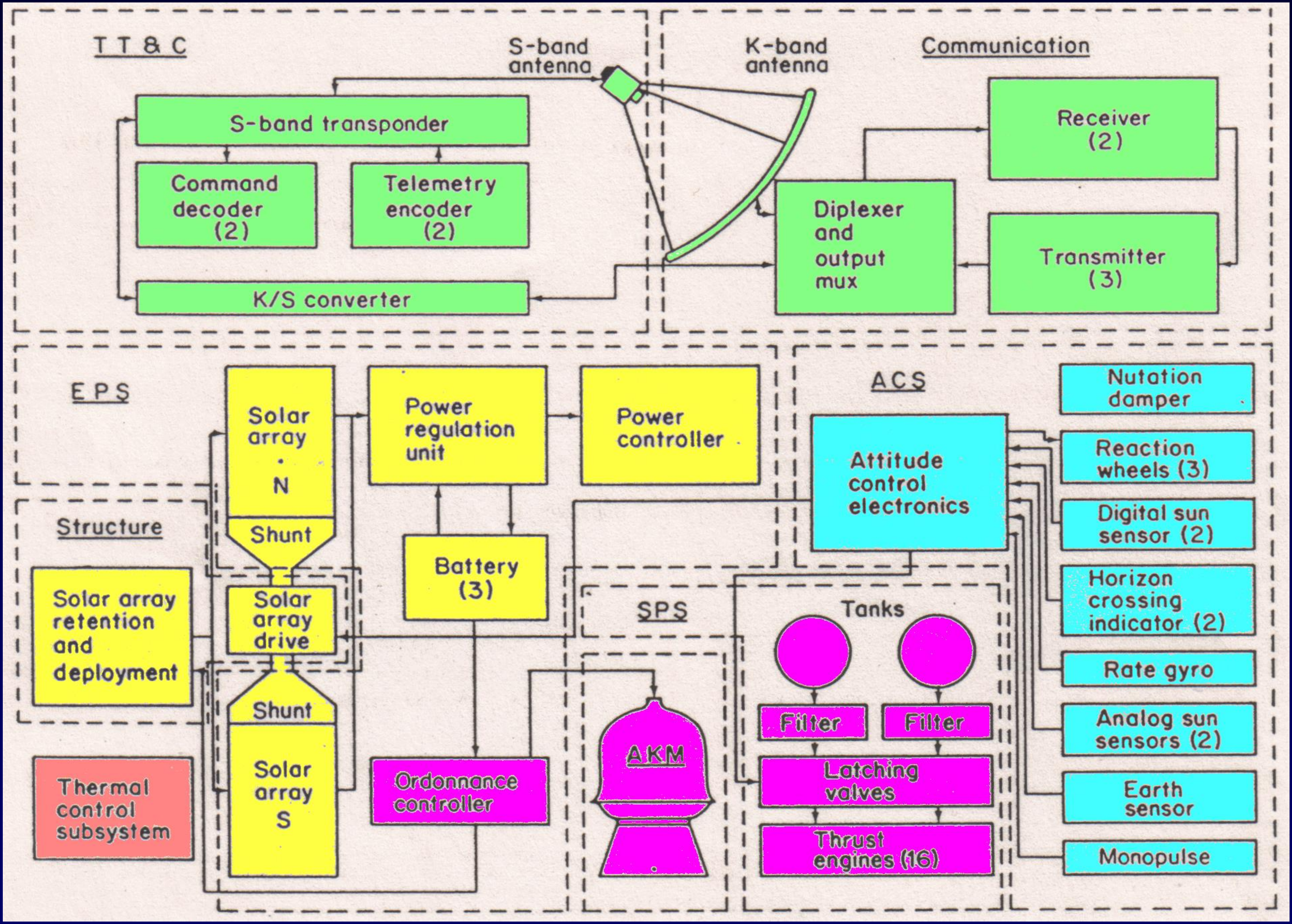
Juno Mission

- The Juno spacecraft launched aboard an Atlas V-551 rocket from Cape Canaveral, Fla., on Aug. 5, 2011, and will reach Jupiter in July 2016.
- The spacecraft will orbit Jupiter 32 times, skimming to within 3,100 miles (5,000 kilometers) above the planet's cloud tops, for approximately one year.
- Juno will be the first solar-powered spacecraft designed by NASA to operate at such a great distance from the sun, thus the surface area of solar panels required to generate adequate power is quite large.
- Three solar panels extend outward from Juno’s hexagonal body, giving the overall spacecraft a span of about 66 feet (20 meters). The solar panels will remain in sunlight continuously from launch through end of mission, except for a few minutes during the Earth flyby.

Juno Solar Power System

- Juno benefits from advances in solar cell design with modern cells that are 50 percent more efficient and radiation tolerant than silicon cells available for space missions 20 years ago.
- The mission's power needs are modest, with science instruments requiring full power for only about six hours out of each 11-day orbit
- With a mission design that avoids any eclipses by Jupiter, minimizes damaging radiation exposure and allows all science measurements to be taken with the solar panels facing the sun, solar power is a perfect fit for Juno.

Summary of Overall Spacecraft Systems



Space Based Infrared Systems (SBIRS), GEO

- Global, persistent missile surveillance continues to be a critical national security space mission.

- A Cold War focus on the strategic ICBM threat has extended to include emerging threats such as short- and mid-range ballistic missiles in the hands of many nations - some openly hostile to the United States and its allies.

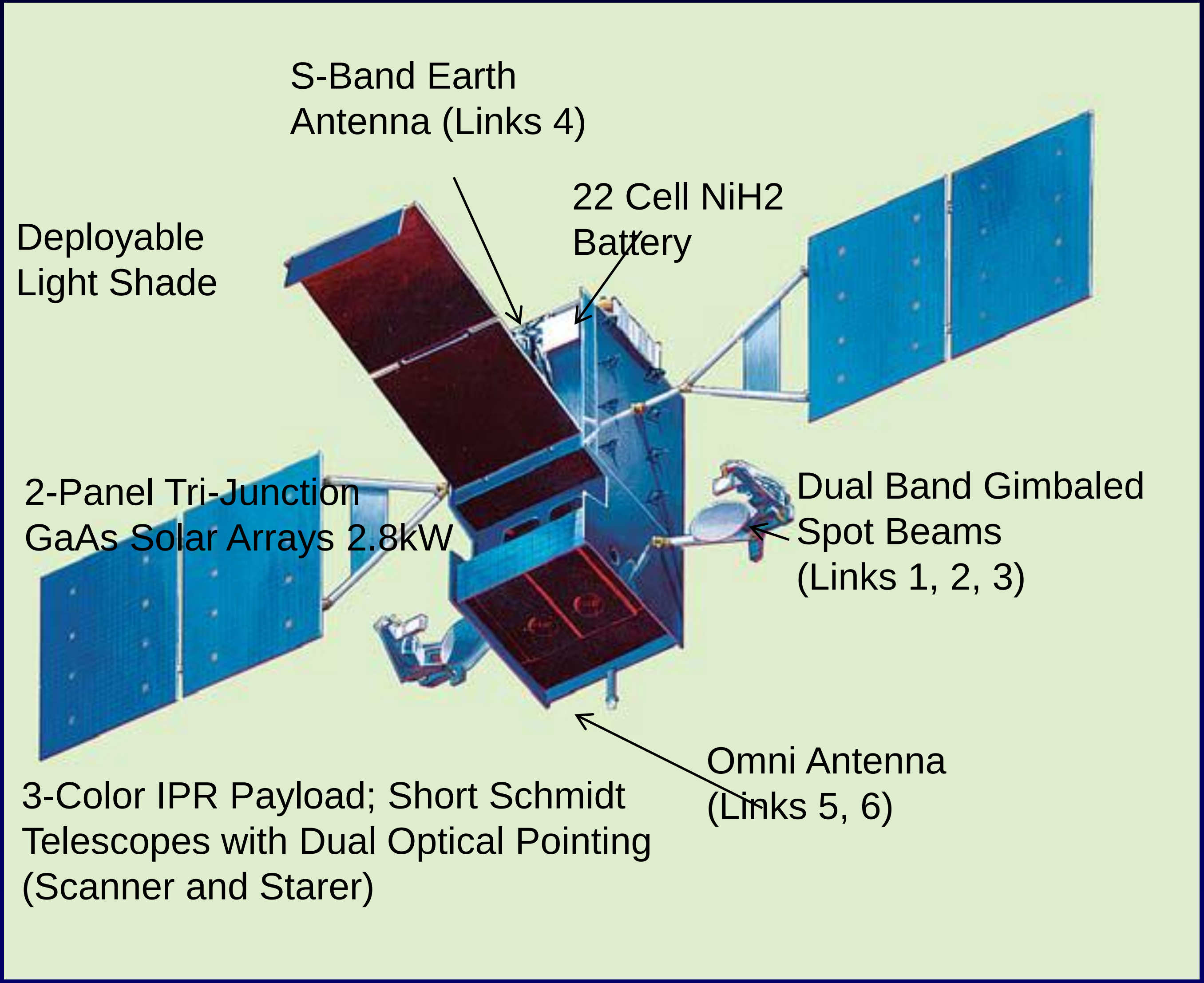
Space Based Infrared Systems (SBIRS) GEO

The U.S. Air Force, in partnership with Lockheed Martin is deploying Space Based Infrared Systems (SBIRS) contributes to the Department of Defence mission to deter war and protect the security of the U.S. by providing timely and accurate missile warning/defence information.

The SBIRS systems are critical for protection against global and theatre ballistic missile attacks against the U.S., its deployed forces and its allies.

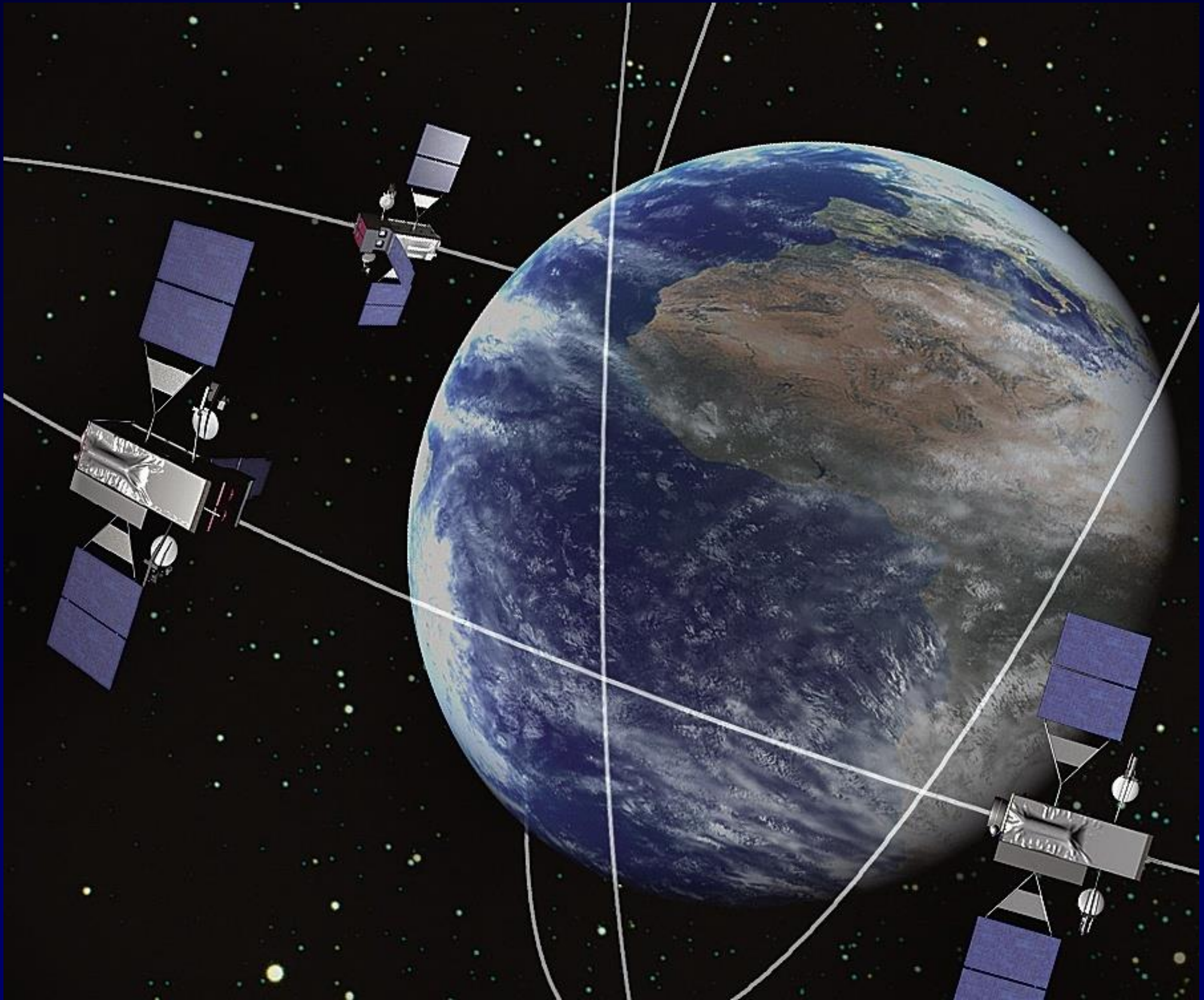
3-axis stabilized with 0.05 deg pointing accuracy

Space Based Infrared Systems (SBIRS)
12 year design life

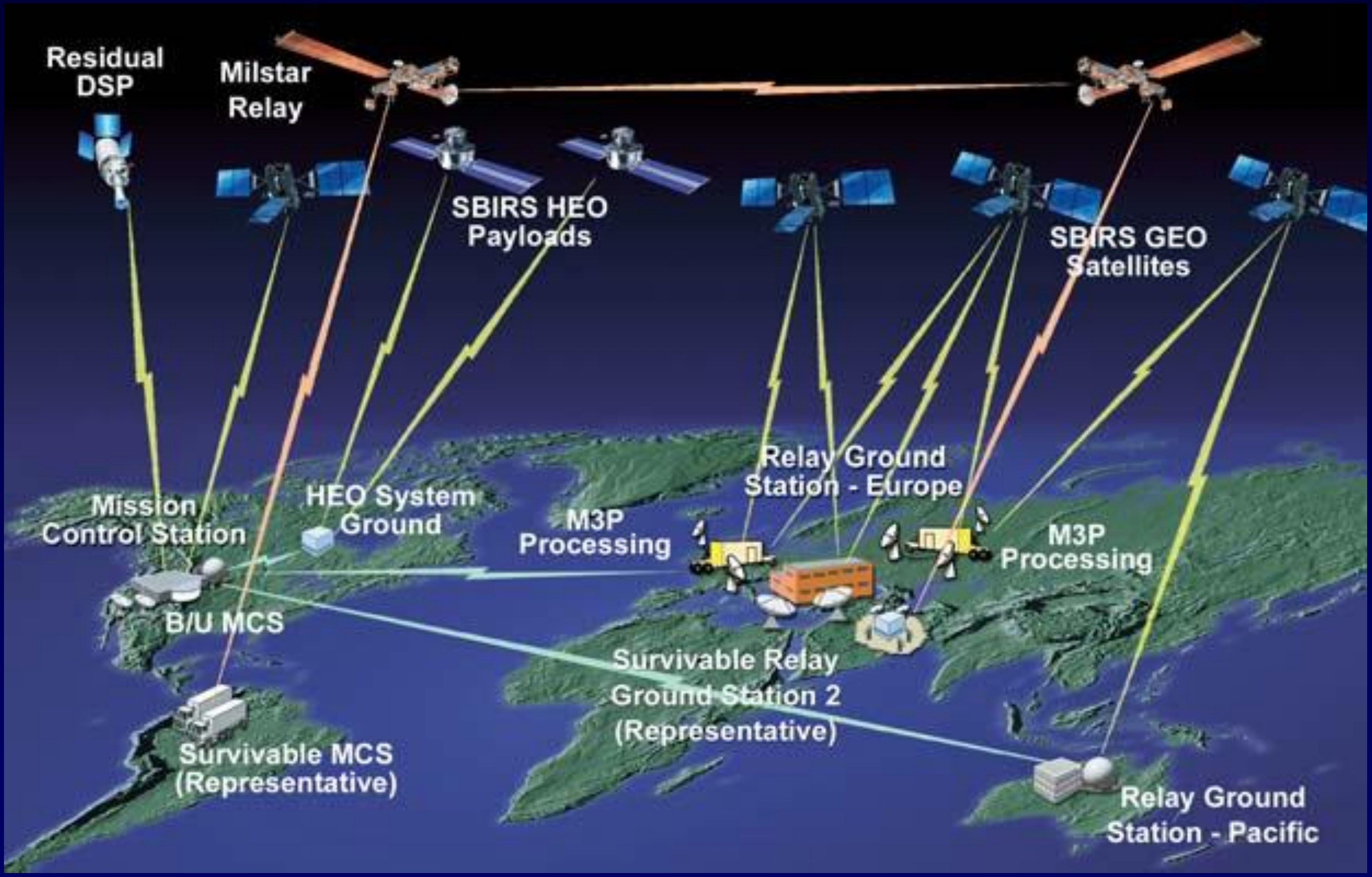


Link Band Function
1-S down Ka Survivable mission data
1-T down Ka Normal mission data
2 up QHF Anti-jam commanding
3 down Ka Wideband sensor data
4 down S Theater mission downlink
5 down S Backup SGLS telemetry downlink
6 up S Backup SGLS commanding

SBIRS - 4 GEO satellites, 2 HEO payloads

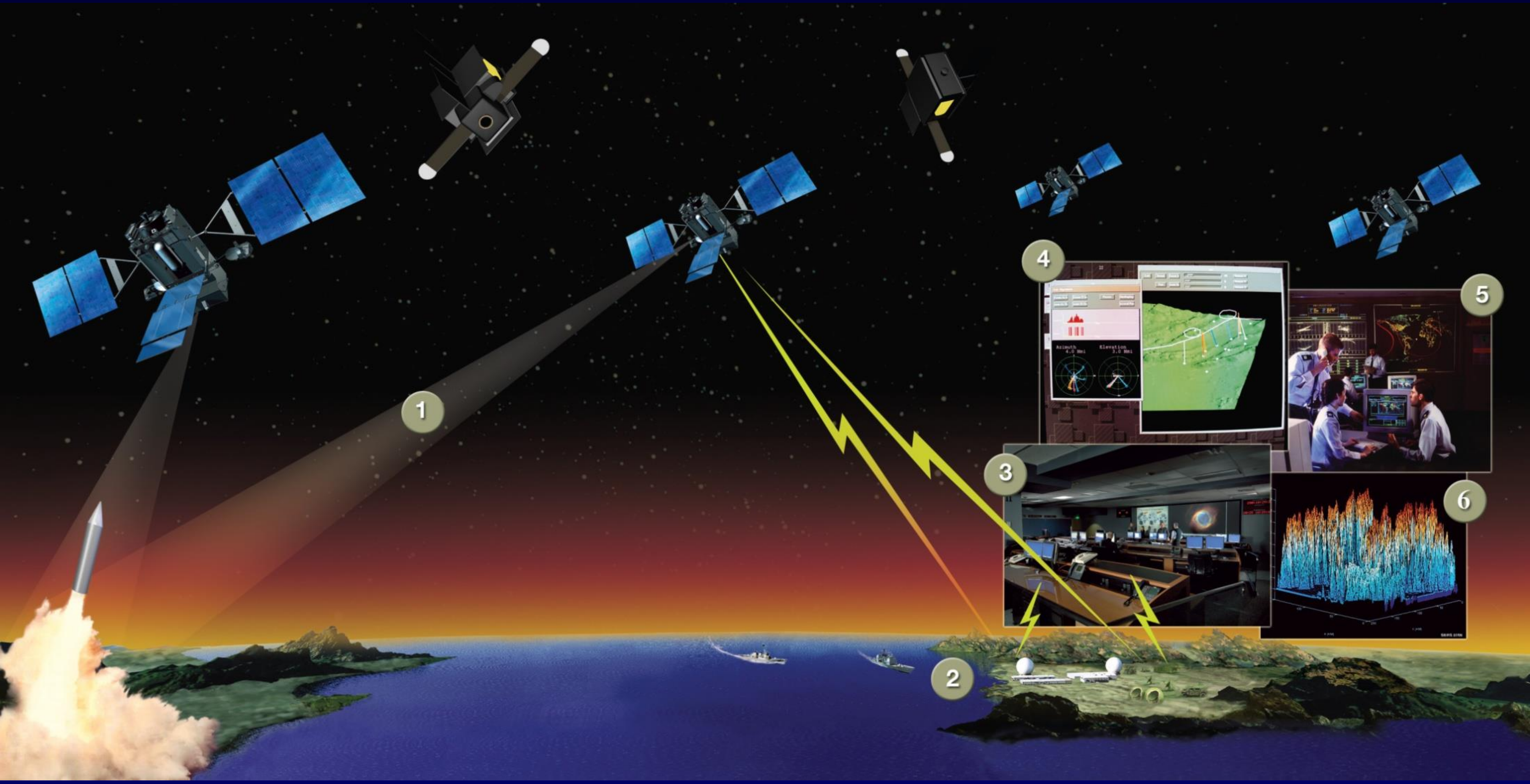


SBIRS System Architecture



Defence Support Program (DSP)

GEO & HEO satellite constellation detecting missile launch



3. The Mission Control Station

5. Air Force Space Command

Ground Station Features

- Manages SBIRS objective constellation of 4 GEO satellites,
2 HEO payloads, and legacy Defence Support Program (DSP) satellites
- Key functions
    - Mission planning/payload tasking
    - Constellation management/TT&C
    - Mission processing
    - Event reporting and data distribution
    - Ground control

LEO (1350 km) infrared observation satellites working in pairs

BOOST PHASE
MIDCOURSE PHASE
RE-ENTRY PHASE

Two Northrop Grumman Space Tracking and Surveillance System (STSS) satellites on-orbit, demonstrating capabilities required for birth-to-death tracking of ballistic missiles and other cold objects in space.

Integrated Detection Systems